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ORBITAL LAUNCH OPERATIONS

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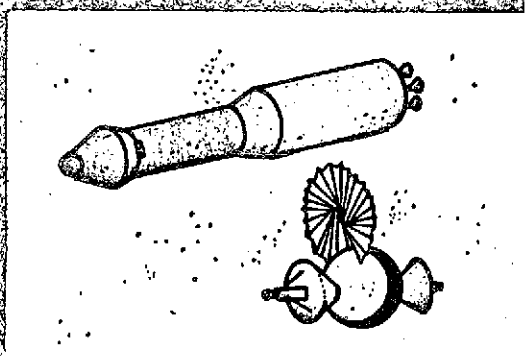
16 JANUARY 1962



ORBITAL LAUNCH OPERATIONS

VOL. III ORBITAL OPERATIONAL TECHNIQUES

VOUGHT ASTRONAUTICS
AMERICAN MACHINE & FOUNDRY CO.
DOUGLAS AIRCRAFT CO.

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Control No. TP 64003
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PHASE II PROGRESS REPORT
(14 June 1961 - 16 January 1962)

Prepared for

George C. Marshall Space Flight Center
(Future Projects Office)
Huntsville, Alabama

Study of

~~ORBITAL~~ LAUNCH OPERATIONS

Volume III

ORBITAL OPERATIONAL TECHNIQUES

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Dallas 22, Texas

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INTRODUCTION

Two major techniques for exploiting the orbital launch operations concept are orbital assembly and propellant transfer. In support of the Orbital Launch Operations Study for NASA by the Chance Vought Corporation, American Machine and Foundry Company was requested to study and evaluate the concepts associated with orbital assembly and Douglas Aircraft Company was requested to perform a similar type study with propellant transfer as a basis. A preliminary investigation of a third technique, dealing with crew transfer, which may be used in conjunction with either of the two major techniques mentioned previously, has been studied by Vought Astronautics.

Part 2 contains the results and recommendations of American Machine and Foundry Company derived from their study of the orbital assembly technique. In support of the recommendations set forth, certain preliminary designs for hardware to accomplish orbital assembly are furnished.

Part 3 reports the results of the study by Douglas Aircraft Company into the various aspects of propellant transfer as an orbital operations technique. In this part, conclusions, recommendations and preliminary designs relating to this technique are discussed.

Part 4 presents a discussion, resulting from a cursory examination, of the various aspects relating to the crew transfer technique. This part presents, on a qualitative basis, certain observations and conclusions. It is intended that the observations be used only as a basis for a subsequent and more detailed investigation of this subject.

PART II

ORBITAL ASSEMBLY

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

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2.1 SUMMARY

The retrieval, docking, and assembly portion of this phase of the study defined and evaluated the problems associated with the assembly of space-vehicles and orbital facilities for launching space-craft from orbit. Techniques, conceptual designs, and specific procedures were generated as possible solutions to these problem areas and resulted in specific hardware concept recommendations for accomplishment of the initial manned lunar mission. A pictorial representation of the recommended system concept is shown in Figure 2-1.

This portion of this phase of the program was divided into the following areas:

1. Orbital Assembly Techniques
2. Retrieval Analysis
3. Docking and Assembly Analysis
4. Use of Man and OLF
5. Weight Summaries
6. Mission Success Probability

The specific results of this portion of the study were:

2.1.1 Orbital Assembly Techniques

An analysis of the large number of possible configurations that potentially might become the Space Vehicle for the initial manned lunar landing and return mission indicated the need for a "universally" applicable system for performing the retrieval, docking, and assembly operations. Both component and systems concepts specifically tailored for a particular configuration have a short life history when configurations are rapidly changing. Based on an examination of 10 concepts (Figures 2-5 thru 2-14) which covered a good portion of the gamut of possibilities, it was possible to determine the operations and tasks which were common to all concepts as well as the special tasks. The inter-relationship of these operations and tasks is shown in Figure 2-4.

A general set of criteria and constraints for the operation of the retrieval, docking, and assembly integrated equipment systems was postulated on the basis of a trade-off examination. These are stated in Section 2.1.1 and served as a guide during both the concepting and evaluation portions of this study.

2.1.2 Retrieval Analysis

As the result of a space mechanics and error analysis, it was possible to perform a trade-off examination and arrive at a logical set of requirements for retrieval operations which is closely coordinated with the rendezvous equipment.

Sixteen retrieval concepts were analyzed during this reporting period. Twelve of these concepts were evaluated against 27 factors for five configurations. The results are shown in Figure 2-22. The concept which consistently rated best was the Rigid Boom-Side Location shown in Figures 2-1, 2-16, and 2-17. Components and operation of this equipment are described in some detail in Section 2.3.6.2.

2.1.3 Docking and Assembly Analysis

This area of the program resulted in the selection of the Conically Arranged Swivel Fastener Assembly as the concept recommended for hardware development. An illustration of this concept is shown in Figure 2-24, and a block diagram of the sequential operations is contained in Figure 2-25.

Among the types of systems considered were common assembly concepts, similar to the recommended method, and special assembly concepts. The special assembly concepts were Propellant Transfer and Crew Transfer. Analysis indicated that these concepts are minor modifications of the common concepts, offering some special advantages, but requiring the development of specialized technologies in addition to those needed for the more basic systems.

2.1.4 Use of Man and OLF

Since the basis for selection was restricted to early accomplishment of the initial manned lunar mission, it was concluded that retrieval, docking, and assembly operations would not require a permanent OLF. The only type of OLF to be considered appeared to be one of a temporary and minimum nature. Such an OLF would be comprised of an Apollo command module (or similar type capsule), with a mission module, outfitted and attached on earth. The mission module would contain assembly, checkout, and launch support equipment.

Four modes of man-machine relationships were investigated: automatic, automatic with man, semi-automatic, and manual. Analysis of these four man-machine modes in light of today's knowledge, and with modest projections in the state-of-the-art, indicates that Automatic with Man operation offers the best probability of mission success. The use of a minimal OLF in conjunction with this mode of operation may enhance mission success.

2.1.5 Weight Summaries

The preliminary weight estimates for the chosen retrieval concept, and docking and assembly concept are:

Rigid Arm - Side Location	740#
Conically Arranged Swivel Fastener Assembly -	2,100#

These values are those required at each interface and are partially provided on each of the two mating modules. The weight values were presented to the program coordinator who integrated them with other sub-system contributions in order to prove the feasibility of the total system with respect to weight effects.

It is worth mentioning that appreciable weight savings, in the form of reduced structure, are gained by using side mounting rather than center mounting for the retrieval & assembly hardware.

2.1.6 Mission Success Probability

A failure analysis was made of each of the sub-systems included in the systems integration and these are shown in the inherent reliability figures given in Table 2-3. The systems concepts were examined for Reliability Growth as shown in Figures 2-33 and 2-34 for the following types of operations: Completely Automatic; Man assisted Automatic; Semi-Automatic; and Manual. The results show that the Man Assisted Automatic system is consistently best, with an indicated Mission Success Probability of 99% in 1967 if the reliability growth rates projected are maintained. Additional curves are presented to show the effect of number of modules assembled on the Mission Success Probability for 1967 for the various systems using the best combination of techniques.

2.1.7 Schedule and Cost Summary

Schedules and Costs are presented for the design, qualification testing, and fabrication of the (1) Rigid Boom Retrieval Mechanism and the (2) Conically Arranged Swivel Fasteners for Docking and Assembly. The total cost for these two programs is approximately $7.6 M, including the cost of 30 production units, with a scheduled delivering 38 months after go ahead.

2.2 ORBITAL ASSEMBLY TECHNIQUES

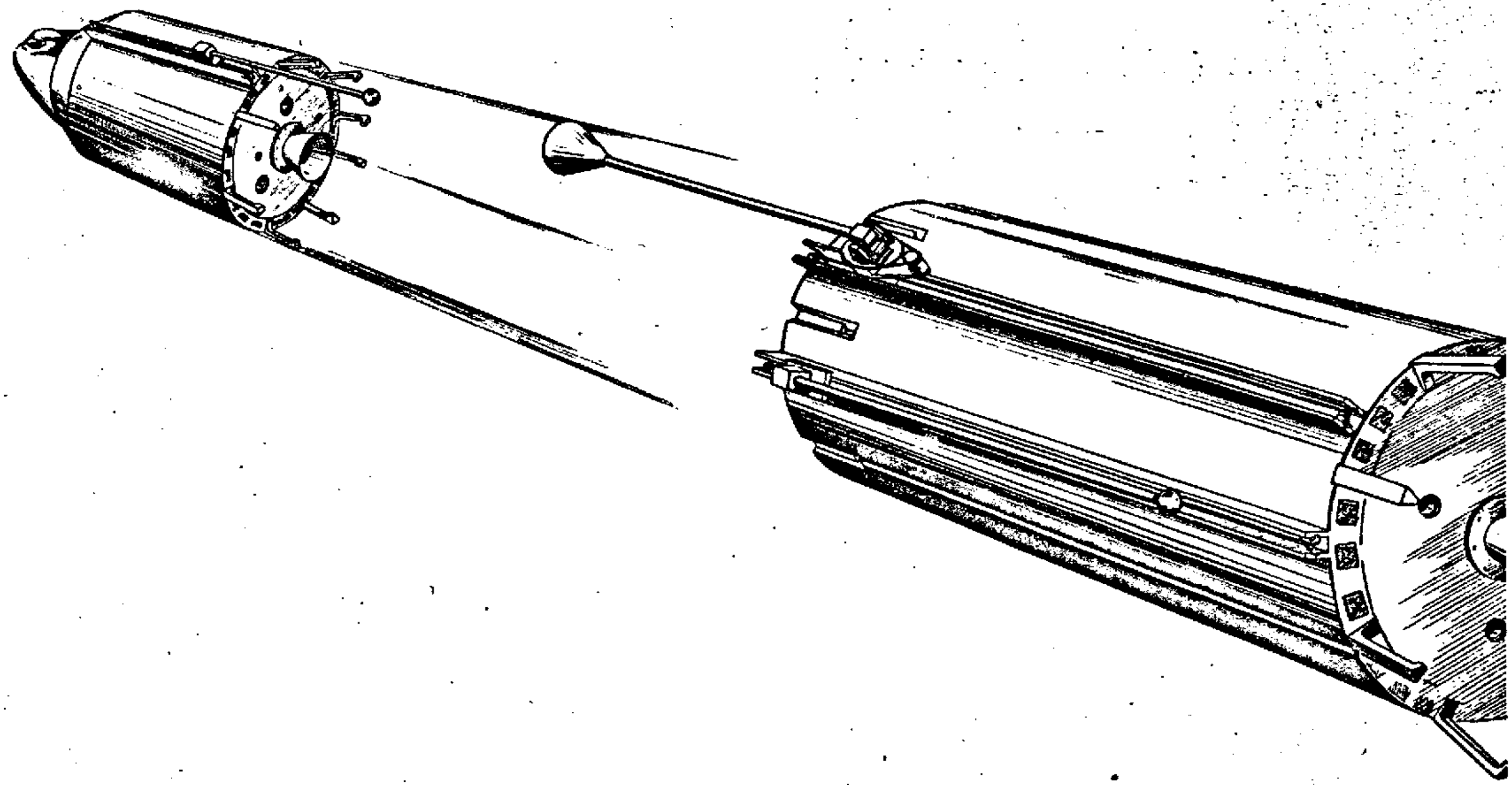
2.2.1 General Considerations

Studies made on Lunar Trajectories by a number of groups (and in particular those performed by NASA at the Lewis Research Center by Weber et al as presented in Technical Note D-866, August 1961) indicate that one of the most decided advantages of orbital departure is that it provides a range of possible assembly orbits, and upon selection of a particular orbit, permits a lunar flight to be initiated from earth twice a day, on any day of the month, within certain limitations. This is quite a different situation from that which governs direct flight from earth. So that, even if it were not required to assemble the Space Vehicle in orbit because of limited booster capabilities, it would be substantially beneficial to utilize a parking orbit anyway. Since this operational mode will prevail in every instance, it behooves us to take full advantage of the opportunity, both for initial and subsequent lunar flights, to obtain the maximum payloads consistent with the state-of-the-art development at the time of launch.

This signifies that orbital assembly of the largest payloads possible will always tend to be indicated, with the only variable being the size of the payload needed consistent with the mission; and as a consequence, the number and type of earth launchings will be a variable within this framework. Since this is the situation that prevails, no elaborate analysis is needed to indicate that the mechanisms utilized for retrieval, docking, assembly, and launch from orbit should be designed so as to be applicable to the large number of situations that can possibly be obtained. If we can succeed in accomplishing this desired goal of "universally" applicable orbital techniques, we should be able to attain maximum reliability (thru repeated usage of identical components) with minimum total development time and cost (since individual development for each and every mission will be eliminated).

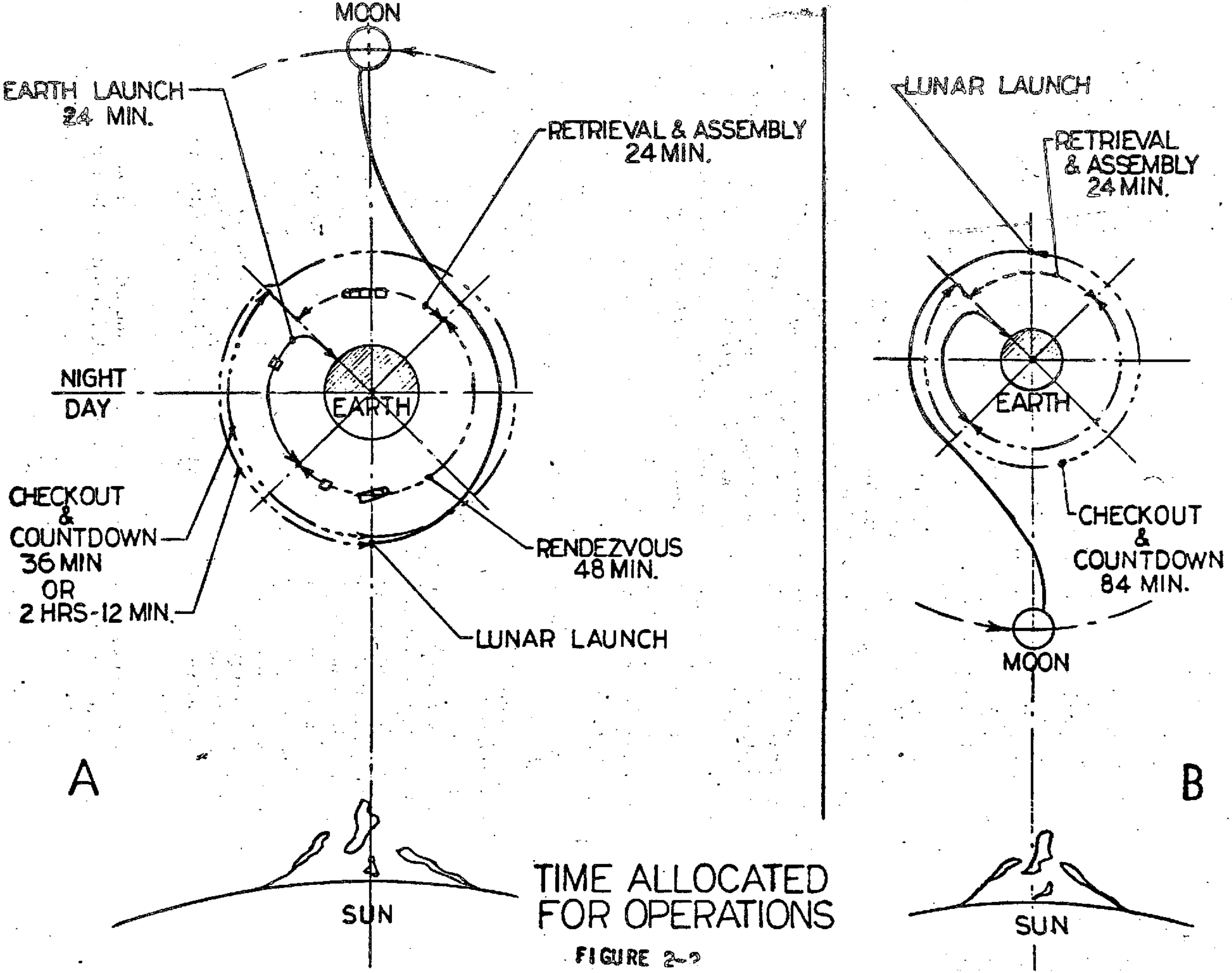
In order to understand the characteristics needed for such "universally" designed assembly equipment, it is necessary that we understand the limitations imposed on orbital departure. The favorable departure times from orbit can be obtained when the parking-orbit plane is oriented to contain the moon at the time of arrival. If this is the case, no costly plane changes need be made and the vehicle would be left in orbit, prior to departure, until it is diametrically opposite the moon. Such a plane can be obtained with an eastward launch from Cape Canaveral twice each day. However, it is quite important that departure from the parking orbit be made at the correct moment, and a delay of several hours or days after the planned instant will mean that the moon will have moved out of the orbital plane. Departure out of the selected plane will incur severe energy penalties or will necessitate long waiting periods in orbit.

In order to enhance the accomplishment of departing from the selected orbit within the realistic time limitations imposed, we restricted our time requirements rather drastically throughout our operations. We assumed that orbital operations will take place at an altitude of approximately 300 International nautical miles, and that the orbital period would therefore be between 90 and 100 minutes. We then restricted the entire process of orbital retrieval, docking, assembly, checkout, and launch to a maximum of two to four orbits depending on the number of modules involved. For the retrieval and assembly operation (for two modules) we restricted the allowable time to 1/4 of an orbit, approximately 24 minutes, and stated that we desired to perform these operations in the shadow of the earth where the thermal and occular effects would be minimized. We then allocated between 36 and 132 minutes to checkout and countdown for the usual situation (and 84 minutes to 3 hrs for the longest situation). Typical operating allocations are shown in Figures 2-2A for the normal case, and in Figure 2-2B for the longest orbital situation. Both of these figures depict the allocations for a two module configuration. Where more than two modules are involved, each individual retrieval and assembly operation would take place in the same quadrant, with the final action as shown in Figure 2-1. The orbital plane selected would therefore be chosen so that



ORBITAL RETRIEVAL & ASSEMBLY TECHNIQUE

Figure 2-1



TIME ALLOCATED
FOR OPERATIONS

FIGURE 2-2

departure would start at the conclusion of the indicated number of required orbits. In the event of a "hold", one additional orbit could be tolerated before the mission might have to be "scrubbed". The necessity of an "aborted" mission would take place only when the velocity penalty exceeded the excess propellant available, or when the life support excess had been used up. While this is an over simplification of the problem, it will suffice as a demonstration of the factors involved.

Earth orientation is one of the simple techniques which would permit orbital retrieval and docking to follow a standard pattern, regardless of the size, shape, or number of modules involved. This is easily accomplished by use of infrared horizon seekers. Accuracy of present day equipment for such stabilization is quoted as ± 1° (by Barnes Engineering as the design specification) and has been found in practice to be ± 5° (Tiros experience as reported by JPL). The advantage of stabilized earth orientation is that the modules always bear the same relationship to each other regardless of position in orbit. While there is a steady thrust cost to attain this, it is quite cheap for the convenience it supplies. Space orientation is cheaper on initial consideration only, but on comparing the accuracies necessary, the effect of accuracy changes, and the complexity of final maneuvers involved, the earth-oriented system appears best for our type of operations and was therefore chosen.

A natural trade-off of equipment complexity will determine the exact distance at which transfer will be made between the control sensors of rendezvous radar and retrieval infrared. A discussion of some of the factors involved is found in the section on Retrieval Analysis. In any event, both systems will utilize the same equipment for obtaining the thrust vectors required after the sensor input switching has taken place. It is not too important for our considerations if the transfer is made at distances up to a mile, but for this study it was assumed that a good compromise would be in the neighborhood of 100 to 500 feet. Similarly, the actual retrieval mechanism could be brought into play at any distance from a few feet up to about 100 feet. For purposes of simplifying the complexity of equipment, a distance of about 50 feet was chosen. The choice of this figure is covered in more depth in the Retrieval Analysis Section. In addition, very close retrieval distances offer the disadvantages that occur due to shock absorption requirements, and the fact that the diameters of the modules being considered are so large (18 to 33 feet) that interferences are possible with only slight orientation errors.

Based on the considerations discussed and other analyses, it was possible to postulate a general set of criteria which served as the initial constraints for this portion of the study.

2.2.1.1 General Criteria and Constraints

(a) Modules are of the following general size and earth weight:

S-IV - 12'10" dia x 41'5" long - 21,500 lbs min.
- 18'4" dia x 41'5" long 113,500 lbs.
S-IV B - 18'4" dia x 75'0" long - 160,000 lbs max.
S-II - 33'0" dia x 63'4" long - 160,000 lbs. max.
S/C - 12'-10" dia. x 27'-6" to 73'-10" lg - Abt 150,000 lbs.

(b) Distance between modules at start of retrieval operation - 500 feet. Distance between modules at point of initial physical retrieval contact - 50 feet.

(c) Relative longitudinal misalignment - ± 5^{o} (10^{o} total).

(d) Relative indexing misalignment - ± 5^{o} (10^{o} total).

(e) Relative axial displacement - 5 feet total.

(f) Relative speed between modules - 0.5 ft/sec max.

(g) Orbital inclination 30^{o} ± 5^{o} with relative inclination difference between modules of 1^{o} max.

(h) Modules are orientated with respect to earth for attitude control.

(i) Total time allocated for retrieval, docking, and assembly - 24 minutes.

(j) Temperature range of parts to be joined will be approximately - 225^{o}F to + 250^{o}F. (dependent on thermal radiation effects only). Temperature of parts mating will be within 100^{o}F differential max.

(k) The simplest, most fool-proof mechanism is the most desired. Weight and cost (of development and testing) are less important than reliability assurance.

(l) No loose parts requiring emplacement are to be considered. All parts used are to be part of the mechanisms attached to the modules.

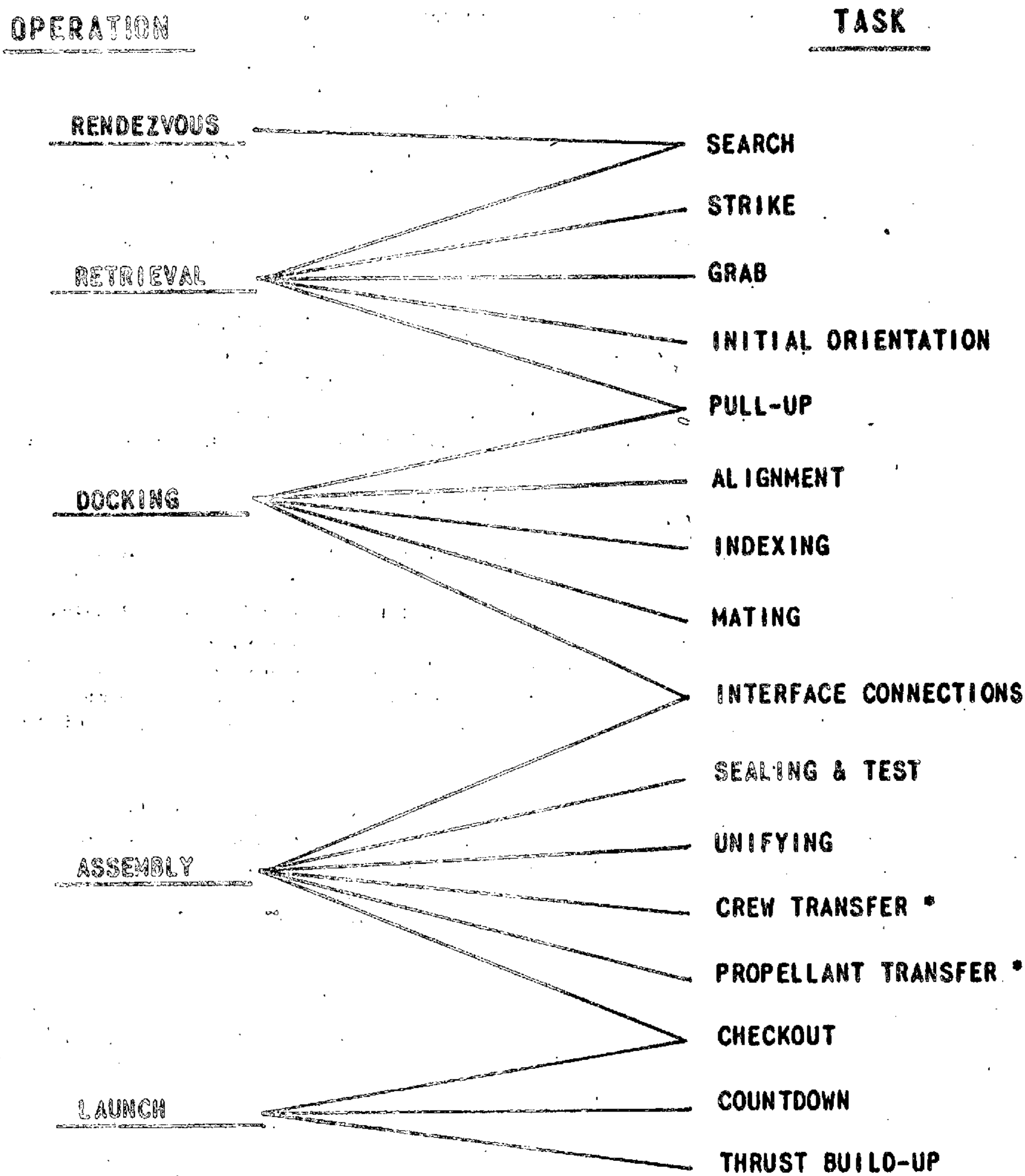
(m) All concepts will have the capability of reuse if necessary to correct for malfunction. The use of integral indicating devices will be incorporated whenever feasible to indicate the satisfactory completion of critical tasks.

2.2.2 Over-all Approach and Plan of Action

Continuing analyses of the factors involved in orbital operations, with special emphasis being placed on "universally" applicable solutions, resulted in a more clearly defined and better organized chart showing the various operations and tasks involved. Figure 2-3 illustrates the relationships between the operations and tasks. Table 2-1 explains the meanings of the terms used as they now apply to OLO. Both the operations and tasks illustrated are common system components of any OLO plan regardless of

ORBITAL LAUNCH VEHICLE (OLV)

OPERATION & TASK BREAKDOWN



* *CREW TRANSFER AND PROPELLANT TRANSFER ARE POSSIBLE TASKS REQUIRED FOR CERTAIN MISSION CONCEPTS AND ARE MERELY VARIATIONS ON THE GENERAL ASSEMBLY SCHEME REQUIRING SPECIAL TECHNOLOGY.*

Figure 2-3

Table 2-1

Definition of Terms for Retrieval, Docking, Assembly, and Launch

A. Retrieval is the process of obtaining a permanent hold on a body in space and effecting a useful connection. It involves a number of tasks, such as:

1. Search which is the concluding phase of rendezvous, often termed fine rendezvous. It involves a controlled homing to the point of contact.
2. Strike which is the initial physical contact between modules. It is characterized by the fact that the objects not only touch each other but are actually caught together.
3. Grab which includes both an extension of the hold established in Strike and the initiation of the positive firm grip which enables all further operations.
4. Initial Orientation of the modules with respect to each other by a positive preplanned sequence originated by the Grab action. This is one of the steps in the progressive reduction in variation of relative position.
5. Pull-up which is the mechanically controlled process that brings the modules into a closer physical relationship thereby allowing a transfer of manipulation to closer tolerance mechanisms. It is the transition phase between the Retrieval and Docking operations.

B. Docking is the process by which two modules are closely orientated to allow positive joining. It involves such tasks as:

1. Alignment where the further reduction in axial angularity and displacement is accomplished to obtain values which will permit subsystem interconnections.
2. Indexing where the further reduction in radial orientation differences is accomplished to obtain values which will permit subsystem interconnections.
3. Mating which is the actual physical joining of the two modules in their required orientation.
4. Interface Connections where the integration of common structural, fluid, and electric systems are made in the required manner to permit the functioning of those services, supplies, and controls shoved between the modules. This is the transition phase between Docking and Assembly.

Table 2-1 (Cont'd)

C. Assembly is the process of so incorporating and integrating the shared portions of the mated modules, that for the length of time they remain assembled, their components act as if they were a single unified package. Assembly is characterized by two types of tasks. One type consists of special cases which need be considered only for certain special mission concepts and involve either or both transfer of crew and propellant. The common tasks which are always involved in Assembly include:

1. Sealing & Test which consists of providing the necessary protection against loss of commodities which must be retained or protection against direct and prolonged exposure to the space environment.
2. Unifying which is the process of joining two modules into a single unified package for the required period of time by providing a redundant hold which will tend to safeguard the operation against the unknowns.
3. Checkout which insures that the proper operation will be attained in flight. In checkout, the condition of both equipment and commodities are inspected. It is mainly concerned with those equipments which had an interface between modules. Checkout provides the transition between Assembly and Launch.

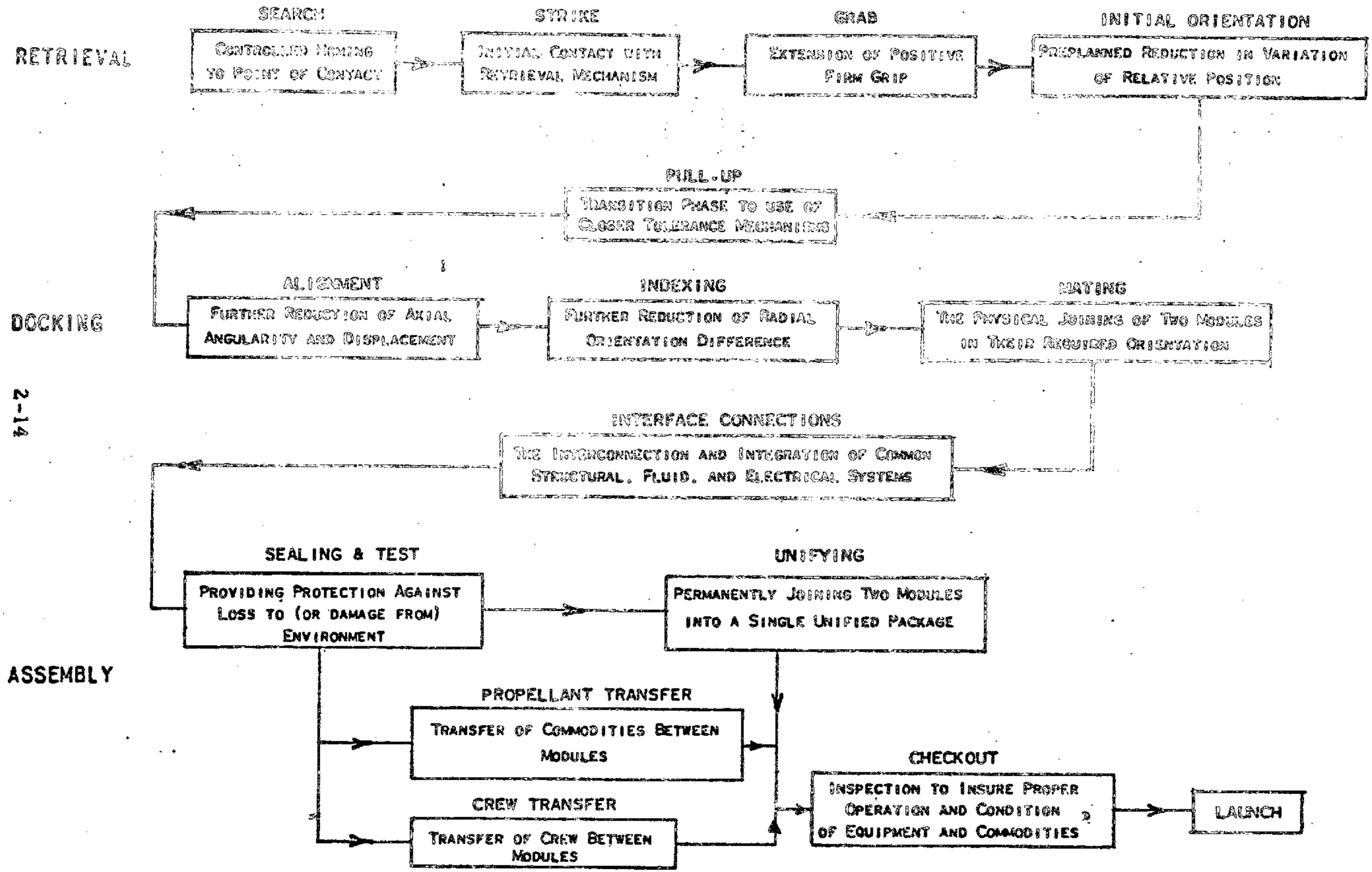
D. Launch is the process which culminates in thrust build-up enabling the Space-Vehicle to successfully depart on its mission. The basic tasks involved in this operation are checkout, countdown, and thrust build-up.

method of approach, technique involved, or number of modules or types used. Depending on the technique or approach taken, emphasis changes and the importance of a task can either increase or disappear, but the essentials appear constant. Within this framework there are some special system components such as Crew Transfer or Propellant Transfer which require technologies and systems of their own. Analysis has shown these special system components are variations on the central theme and still require most, if not all, of the common system components. The inter-relationship between the various task sequences is shown in block diagram form in Figure 2-4.

The possible variations of number and types of modules to be assembled is quite large and is entirely dependent on the latest speculation or information available. Some of the various configurations which have been considered during this study period are shown in block diagram form in Figures 2-5 thru 2-14. As can be readily understood, any retrieval-assembly concept which is sensitive to such variations in configuration will have only limited application and quite possibly only a limited history of study and development before being discarded. The idea of "universality" of application therefore became essential if this portion of the study was to have any meaning for the future.

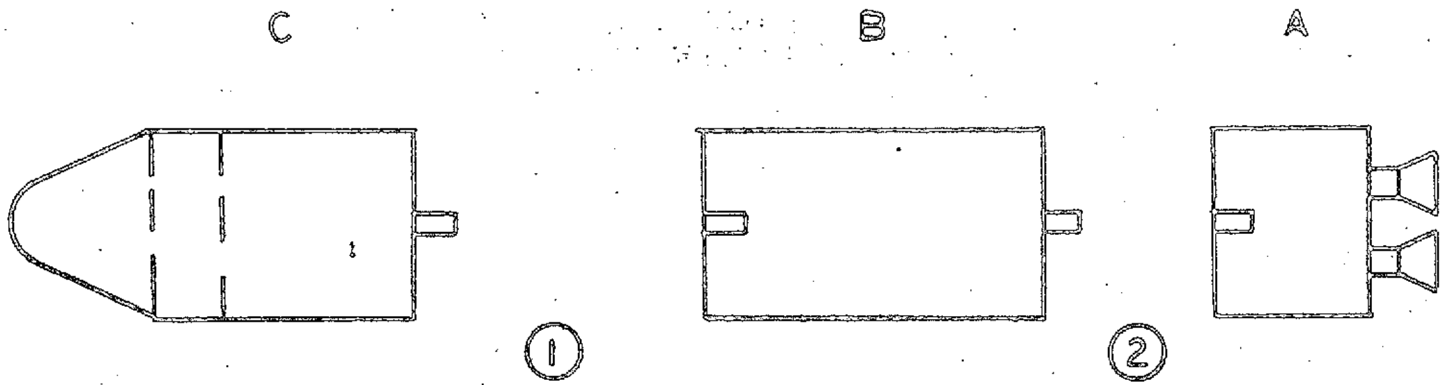
It also became apparent that a continuum of effort is now needed to conceive and develop a system approach to the entire problem rather than spend time on the variety of possible methods and equipments. The previous approach had been more or less on a handbook type basis where the emphasis had been on theoritical examination of all the potentially useful concepts for the individual components without regard to system organization or requirements. The time will come when this handbook approach will again be necessary, but for the present we have a broad enough foundation and understanding to undertake putting together a workable system. Sophistication and optimization of the individual components can wait development needs of the selected system.

The plan of action therefore became one of examining the more universally applicable subsystems required for the operations involved in retrieval, docking, and assembly. After this first phase of concepting was completed, an evaluation was made both of the individual concepts and the systems utilizing the better concepts. The coherence of the whole approach to the over-all problem then became the dominant objective and what emerged is a workable system which satisfies the criteria initially postulated. It is really unimportant how many systems can be derived or even if the system solution proposed is the best of all such possible systems. What is important is that we have a complete approach which can be studied in detail, developed within the required time period, and tested and proved reliable in conjunction with programs now in progress. The need for further paper studies and optimization evaluations has greatly diminished for the present, what is now needed is hardware which can be tested in the working environment and thereby furnish us with information and clues on which to base realisitic evaluations and decisions.



INTER-RELATIONSHIP OF TASK SEQUENCES

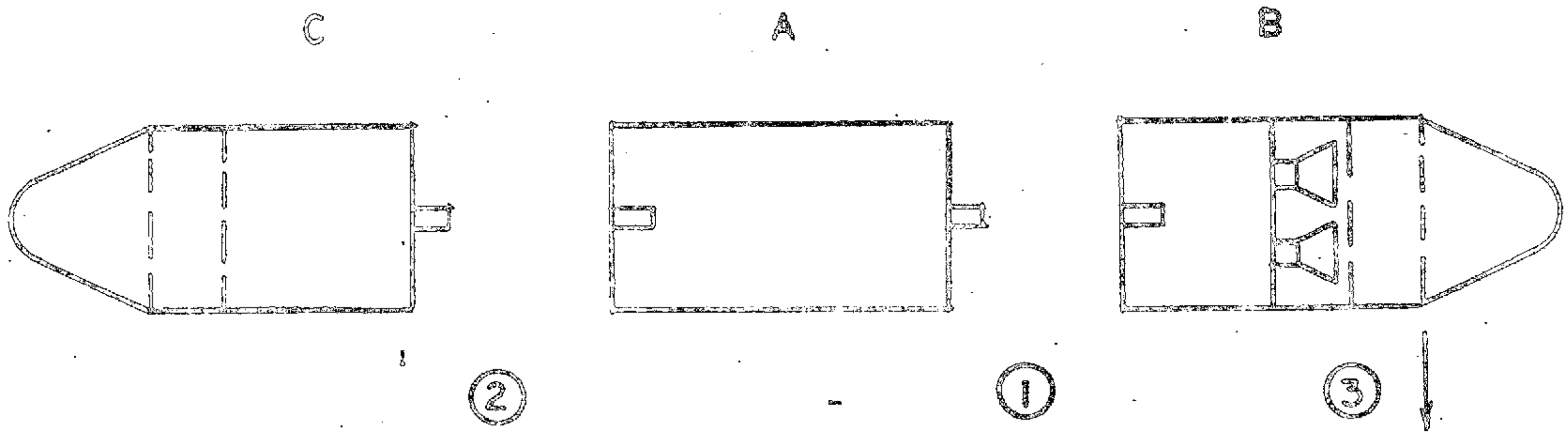
Figure 2-4



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY
2. TWO RETRIEVAL AND ASSEMBLY OPERATIONS
3. NO OLF PROVIDED
4. NO CREW TRANSFER
5. ONLY ONE CREW AND ONE CAPSULE, CREW ORBITED BY C-4
6. MISSION LIFE SUPPORT REQUIRED IN ORBIT
7. THREE LAUNCHES FROM EARTH: ONE C-1 AND TWO C-4

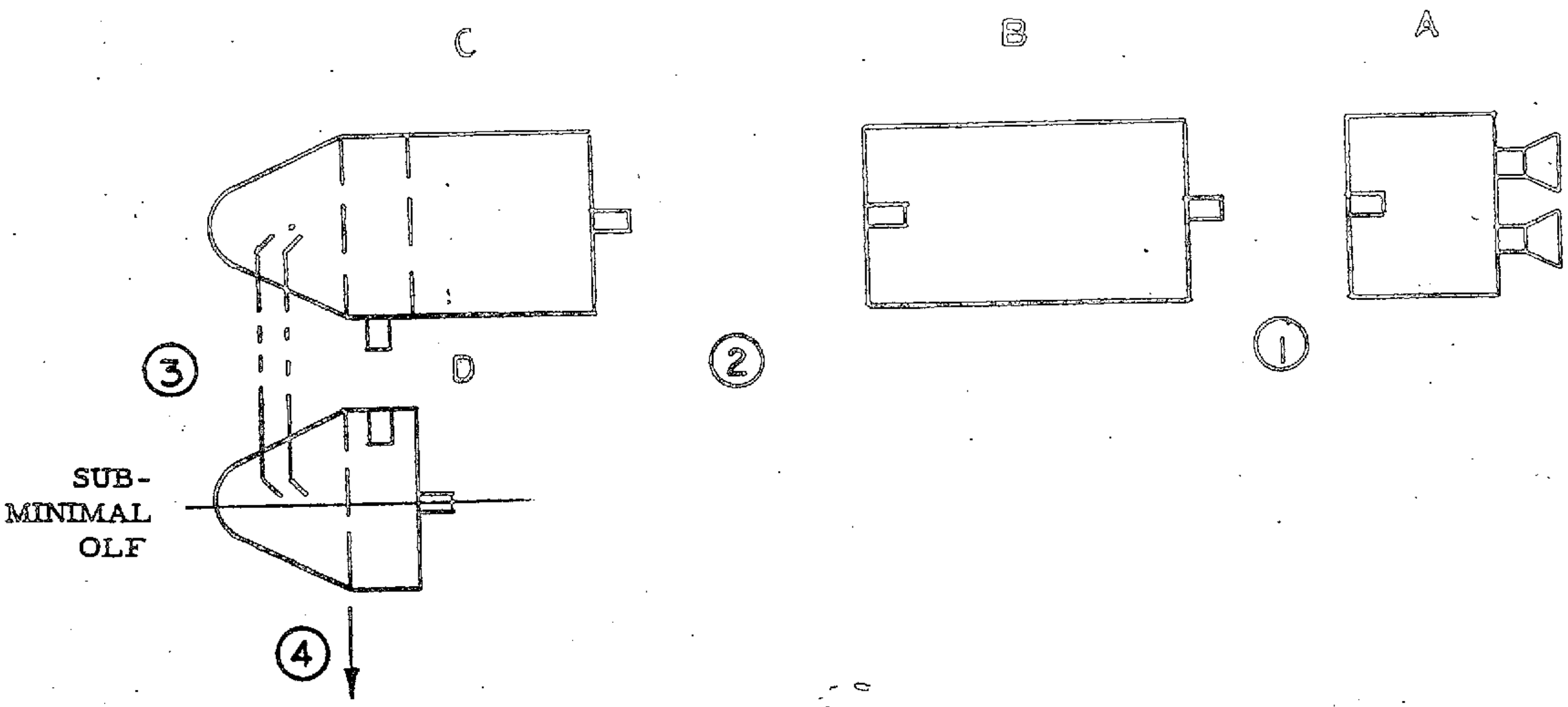
ACCELERATED PROGRAM
CONCEPT #1A
C-4 MAN ASSISTED RETRIEVAL
FIGURE 2-5



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY
2. TWO RETRIEVALS AND ASSEMBLY OPERATIONS
3. INTEGRAL MINIMAL OLF PROVIDED WITH SEPARATION REQUIRED
4. NO CREW TRANSFER
5. TWO CREWS, BOTH ORBITED BY C-4
6. THREE C-4 LAUNCHINGS FROM EARTH

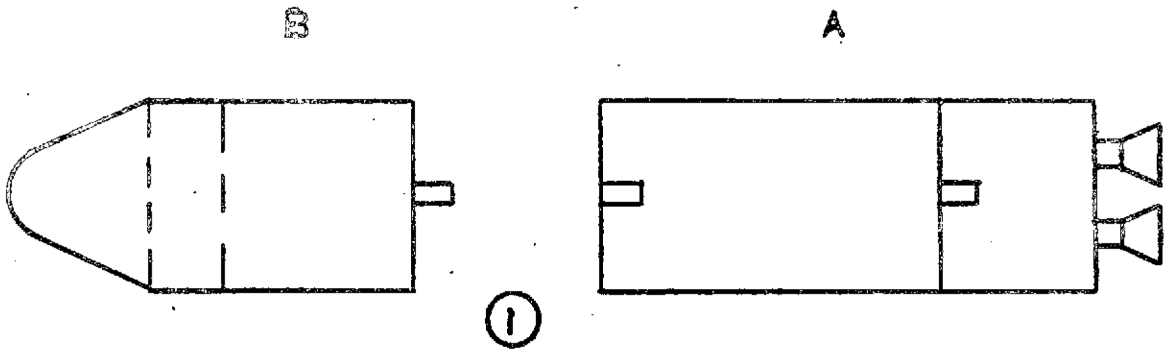
ACCELERATED PROGRAM
CONCEPT 2A
MAN-ASSISTED RETRIEVAL WITH INTEGRAL MINIMAL OLF
FIGURE 2-6



CHARACTERISTICS

1. AUTOMATIC RETRIEVAL AND ASSEMBLY
2. THREE RETRIEVALS, TWO ASSEMBLY OPERATIONS
3. SUB-MINIMAL OLF PROVIDED (CREW TRANSFER AND LAUNCH SUPPORT)
4. ONLY ONE CREW. CREW ORBITED BY C-1
5. CREW TRANSFER
6. NO MISSION LIFE SUPPORT USED WHILE IN ORBIT
7. TWO CAPSULES
8. FOUR LAUNCHES FROM EARTH: TWO C-1 AND TWO C-4
9. CREW ORBITED AFTER SPACE-VEHICLE ASSEMBLY

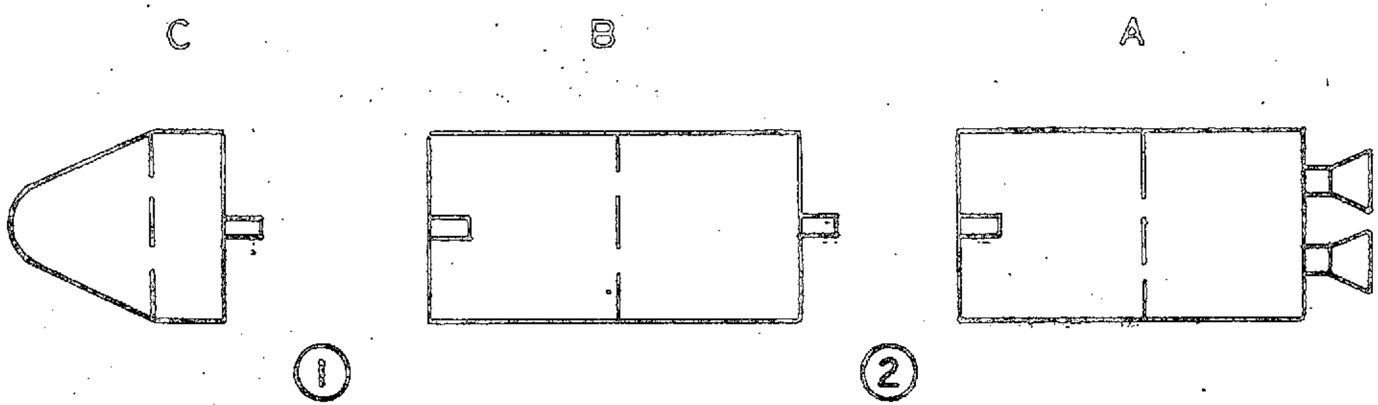
ACCELERATED PROGRAM
CONCEPT #3A
AUTOMATIC WITH CREW TRANSFER
FIGURE 2-7



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY
2. ONLY ONE RETRIEVAL AND ASSEMBLY OPERATION
3. C-5 TYPE BOOSTER REQUIRED FOR OLV
4. CREW ORBITED BY C-4
5. TWO LAUNCHES FROM EARTH: ONE C-4 AND ONE C-5

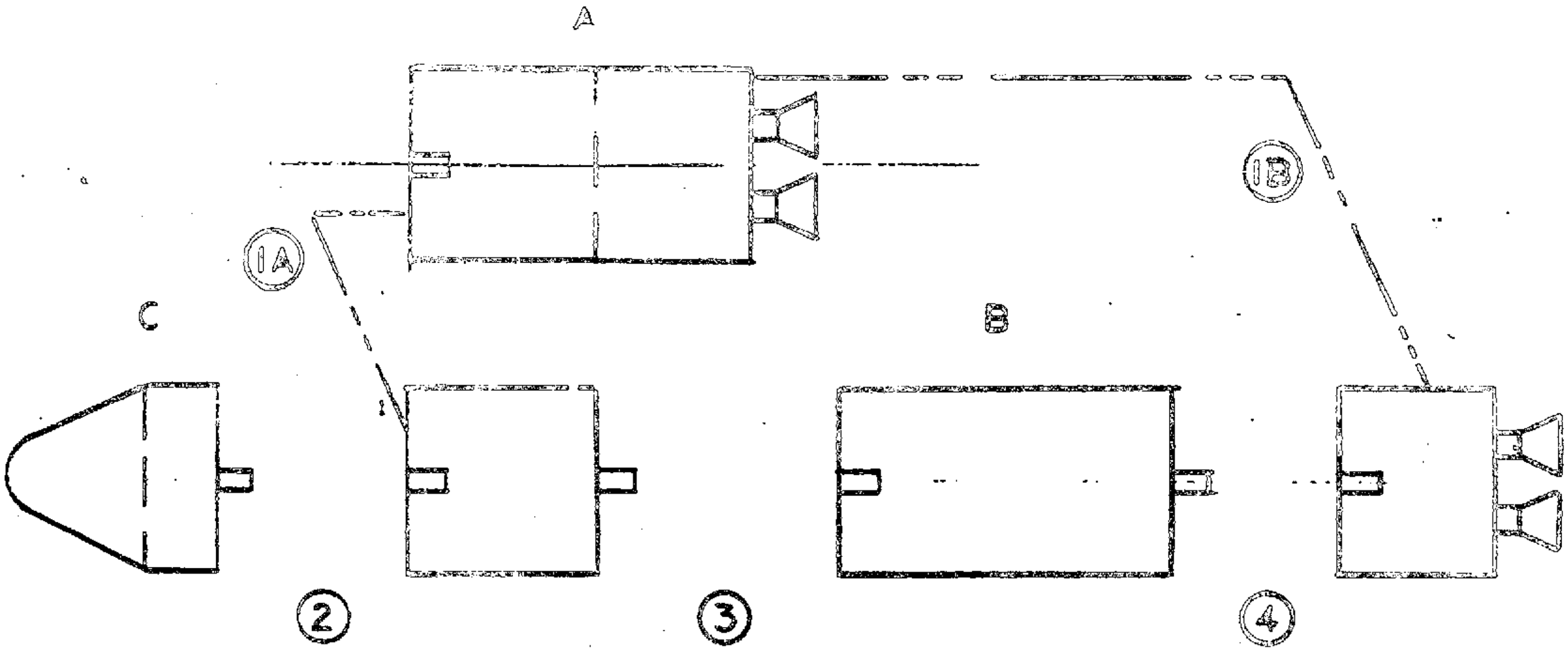
ACCELERATED PROGRAM
CONCEPT #4A
MAN ASSISTED RETRIEVAL WITH C-5 BOOSTER
FIGURE 2-8



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY OR AUTOMATIC
2. TWO RETRIEVAL AND ASSEMBLY OPERATIONS
3. NEW MODULES REQUIRED FOR PORTION OF SPACECRAFT AND OLV
4. CREW ORBITED BY C-1
5. THREE LAUNCHES FROM EARTH: ONE C-1 AND TWO C-4

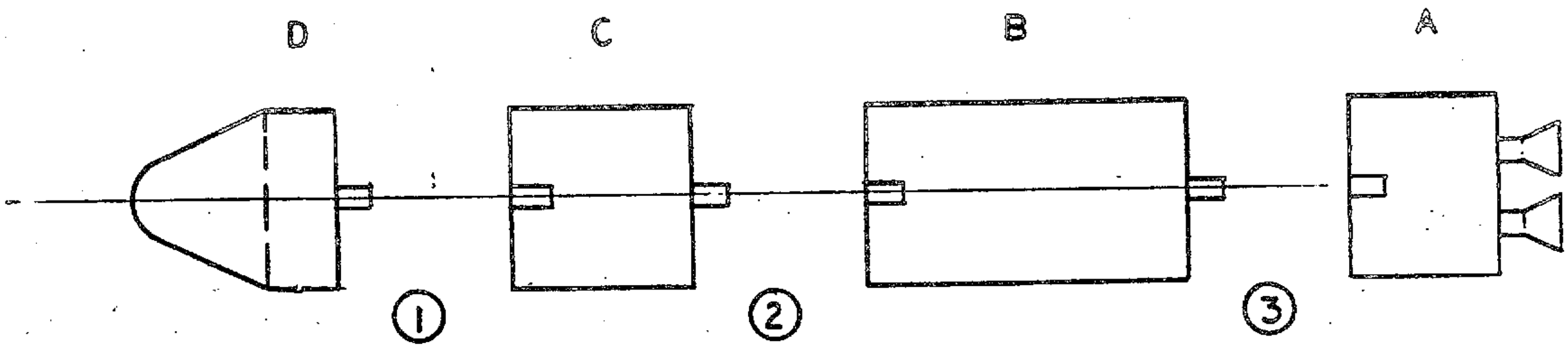
NEW MODULE PROGRAM
CONCEPT #1P
C-1 MAN ASSISTED RETRIEVAL
FIGURE 2-9



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY OR AUTOMATIC
2. THREE RETRIEVAL AND ASSEMBLY OPERATIONS
3. MODULE SEPARATION REQUIRED IN ORBIT
4. SPACE CRAFT FORMED IN ORBIT
5. CREW ORBITED BY C-1
6. THREE LAUNCHES FROM EARTH: ONE C-1 & TWO C-4

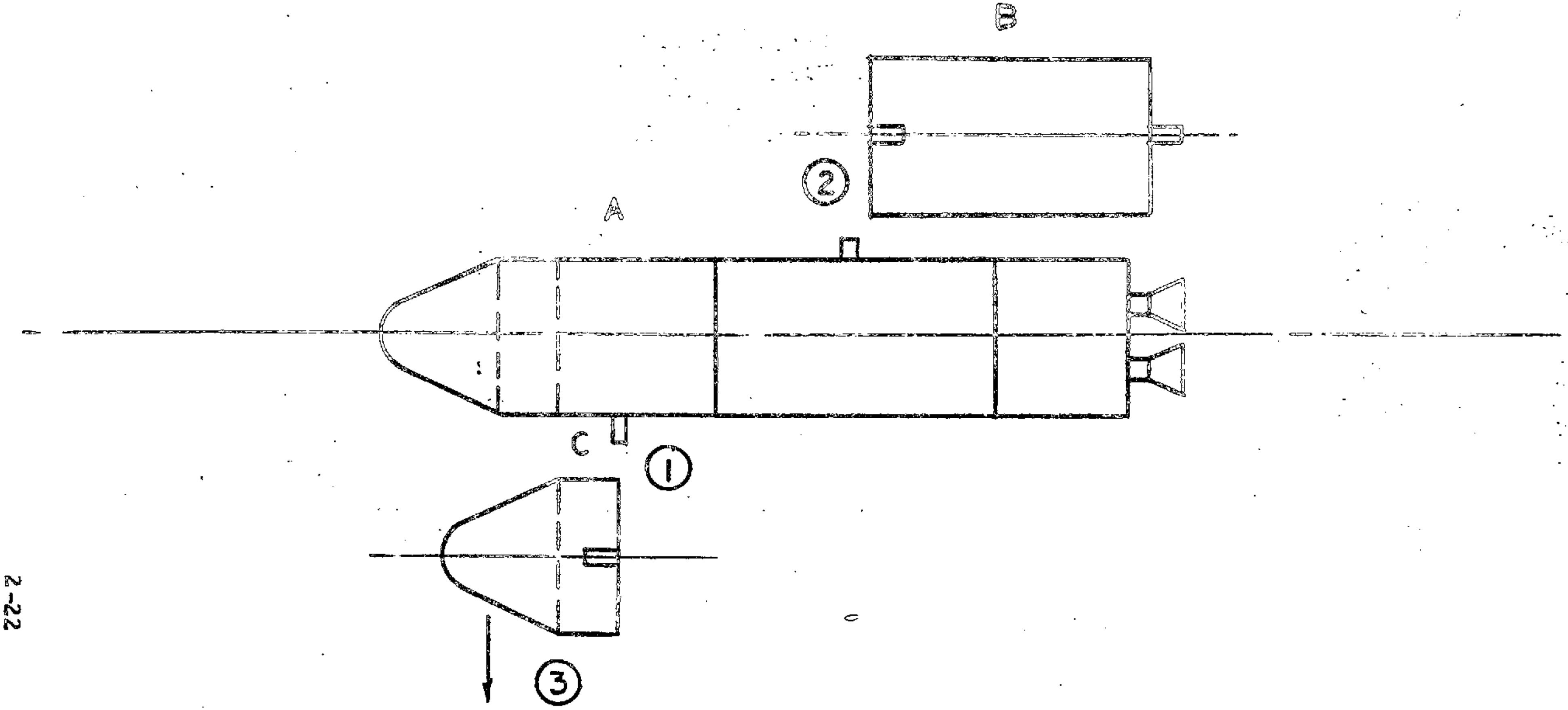
NORMAL PROGRAM-SPECIAL
CONCEPT #1S
SEPARATED MODULE
FIGURE 2 - 10



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY OR AUTOMATIC
2. THREE RETRIEVAL AND ASSEMBLY OPERATIONS
3. SPACECRAFT FORMED IN ORBIT
4. CREW ORBITED BY C-1
5. FOUR LAUNCHES FROM EARTH: TWO C-1 AND TWO C-4

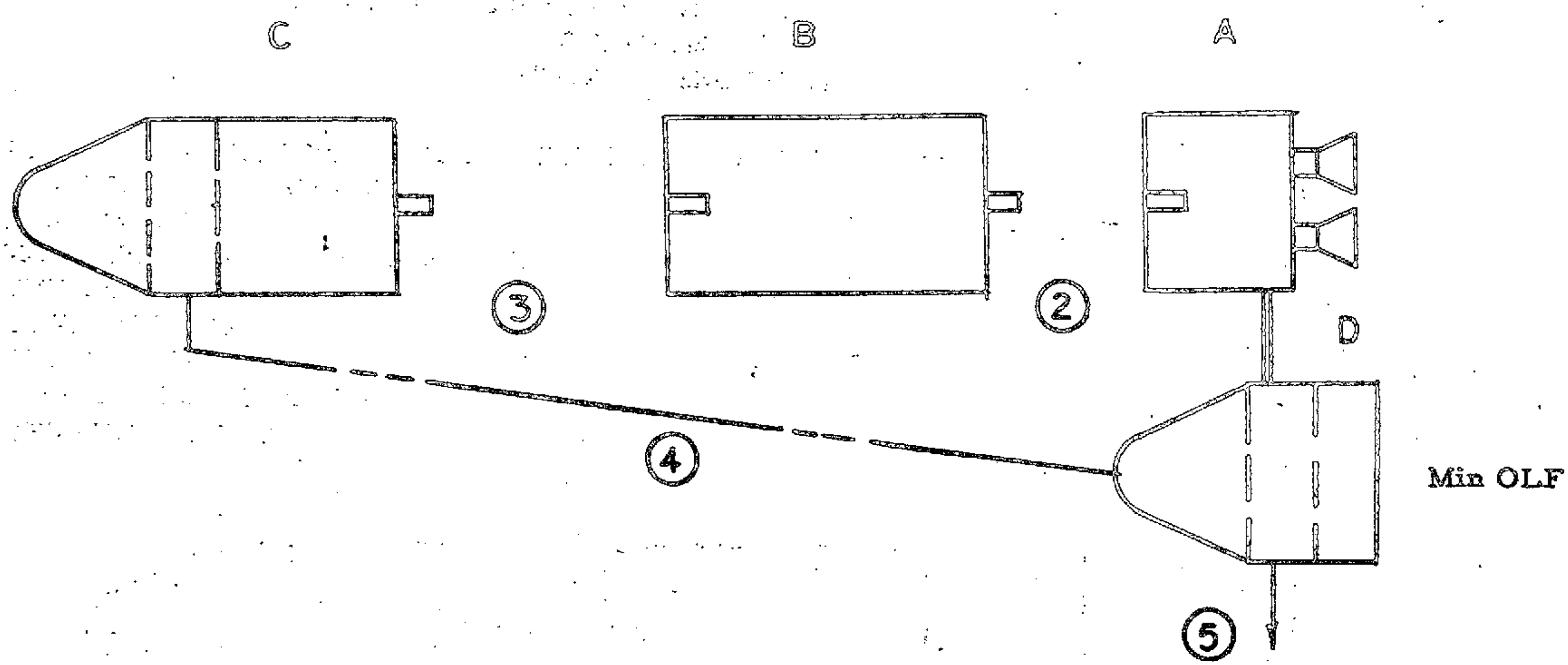
NORMAL PROGRAM
CONCEPT #1
SPLIT SPACECRAFT
FIGURE 2-11



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL
2. SPACECRAFT IS ASSEMBLED ON EARTH
3. CREW TRANSFER AND PROPELLANT TRANSFER REQUIRED
4. TWO CAPSULES BUT ONLY ONE CREW - ORBITED BY C-1
5. MINIMAL OLF POTENTIAL
6. THREE LAUNCHES FROM EARTH: ONE C-1 AND TWO C-4

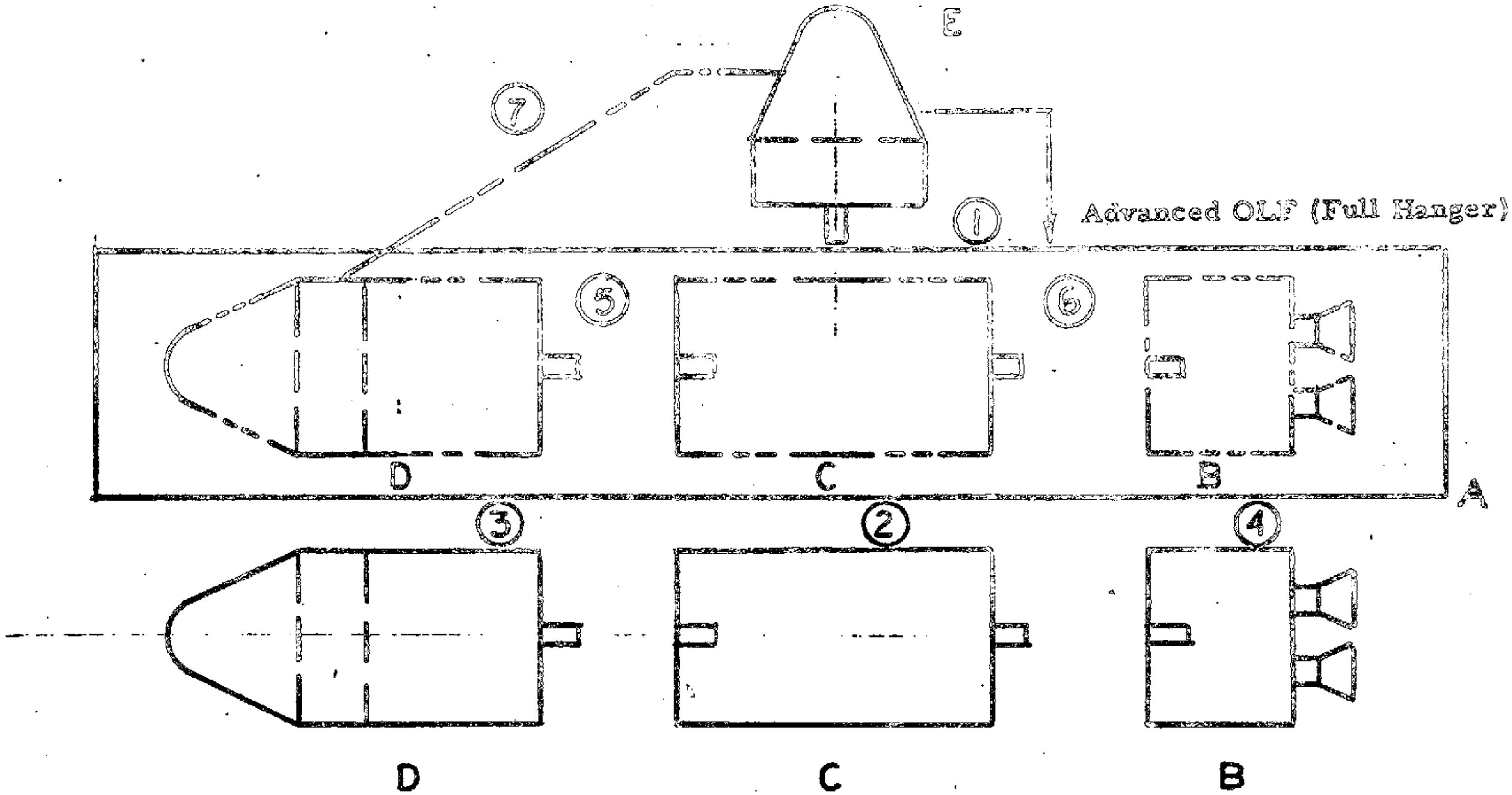
NORMAL PROGRAM
CONCEPT #2
PROPELLANT AND CREW TRANSFER
FIGURE 2-12



CHARACTERISTICS

1. MAN ASSISTED RETRIEVAL AND ASSEMBLY
2. THREE RETRIEVAL AND ASSEMBLY OPERATIONS
3. MINIMAL OLF PROVIDED. NO MISSION LIFE SUPPORT USED.
4. CREW TRANSFER REQUIRED
5. CREW ORBITED BY C-1, TWO CAPSULES REQUIRED
6. FOUR LAUNCHES FROM EARTH: TWO C-1 AND TWO C-4

NORMAL PROGRAM
CONCEPT #3
MINIMAL OLF AND CREW TRANSFER
FIGURE 2-13



CHARACTERISTICS

1. MANNED RETRIEVAL AND ASSEMBLY
2. OLF PROVIDED (SHIRTSLEEVE ASSEMBLY, CREW TRANSFER, REPAIR & LAUNCH SUPPORT)
3. CREW TRANSFER
4. ONLY ONE CREW
5. NO MISSION LIFE SUPPORT USED
6. TWO CAPSULES
7. OLF REMAINS IN ORBIT
8. FIVE LAUNCHES FROM EARTH: THREE C-1 AND TWO C-4

NORMAL PROGRAM
CONCEPT #4
MANUAL WITH ADVANCED OLF (FULL HANGER)
FIGURE 2-14

2.3 RETRIEVAL TECHNIQUE

2.3.1 Introduction

This section describes concepts and methods that could be used to mate two modules in orbit that are initially separated by a distance of at most a few hundred feet. In addition, the limitations of flight control techniques are discussed. It is these restrictions which require that the final retrieval and joining (at a separation distance of about 50 feet) be accomplished by physically reaching out and grasping the payload package. The capabilities of flight control are interrelated with space mechanics effects, therefore those applicable areas of orbital mechanics were investigated.

The basic philosophy that permeated the selection of concepts, systems, and mechanisms for retrieval was that they be universally adaptable to any permutations in orbital space vehicle configuration such as might occur with various numbers of the different Saturn payloads (see Figures 2-5 through 2-14).

The discussion of retrieval techniques that follows encompases these major topics:

(a) Retrieval Events
(b) Limitations of Thrusting Control Techniques
(c) Applicable Orbital Mechanics
(d) Search Operation
(e) Mechanical Retrieval Techniques
(f) Evaluation of Mechanical Retrieval Techniques

2.3.2 Retrieval Events

The sequence of events which result in the final joining of two modules through the use of physical retrieval involve: search, strike, grab and initial orientation. This sequence is preceded by both gross rendezvous phase and intermediate rendezvous operations. The terminal or fine rendezvous phase is the starting point for the search operation. Gross rendezvous is considered to bring the chaser vehicle to within 25 miles of the target module. The intermediate rendezvous phase closes the gap between the two vehicles to within 500 feet or less. The fine terminal rendezvous takes the chaser vehicle to within 50 feet of its target, at which point physical attachment can be made. The choice of the distance at which physical attachment begins is substantiated elsewhere in this section.

2.3.3 Limitations of Thrusting Control Retrieval Techniques

The choice of a physical technique for achieving rendezvous within the last 50 feet of separation was based upon the uncertainties associated with flight control maneuvers. It is at this terminal point in the retrieval program where the utmost reliability is required to prevent washing out the mission due to colliding of vehicles without the capabilities of impact load attenuation.

Some of the limitations that exist with flight control retrieval techniques are as follows:

(a) The uncertainties in range and rate measurements due to inaccuracies in sensors and instrumentation.

(b) The inaccuracies of the orbital equations fed into computers due to the assumptions made in their development.

(c) The round off errors associated with the use of digital computers. The weights of such equipment becomes excessive when all higher order terms are to be considered.

(d) The inaccuracies obtained in the control of rocket thrust cutoff points even when vernier jets are used.

(e) The minimum sensitivities and accuracies of thrust control jets.

(f) The dead band or limits of resolution associated with the use of servo equipment.

(g) The errors inherent in orbital ellipticities due to failure to enter perfectly circular orbits, also the error caused by being slightly out of proper plane.

(h) The inability to sense or predict satellite perturbations due to earth eccentricities.

There are probably many more sources of error to which flight control techniques are susceptible. The quantitative analysis of these errors is an area that requires a study in itself. It appears evident from the cursory look given to this problem that the uncertainties of flight control techniques eliminate its use for the terminal retrieval phase at least for the time period under consideration.

2.3.4 Applicable Orbital Mechanics

In order to determine the effect of orbital mechanics on flight control maneuvers, equations and solutions to these equations were developed using the perturbation technique to obtain the change in velocity that is required in order to achieve rendezvous when separation distances X and Y are shown.

The resulting solutions which illustrate the relations between X, Y and $\dot{X}$, $\dot{Y}$ are as follows:

$$(1) \quad Y = 2\,\frac{\Delta\dot{X}}{W_a}\left[1 - \frac{1}{\sqrt{\left(\frac{\Delta\dot{Y}}{2\Delta\dot{X}}\right)^2 + 1}}\right] + \frac{\Delta\dot{Y}}{W_a}\,\frac{1}{\sqrt{1+\left(\frac{2\Delta\dot{X}}{\Delta\dot{Y}}\right)^2}}$$

$$(2) \quad X = 3\left(\frac{Y}{2} - \frac{\Delta\dot{X}}{W_a}\right) \text{Arc Tan}\left(\frac{-\Delta\dot{Y}}{2\Delta\dot{X}}\right) + \frac{4\Delta\dot{X}}{W_a}\,\frac{1}{\sqrt{1+\left(\frac{2\Delta\dot{X}}{\Delta\dot{Y}}\right)^2}}$$

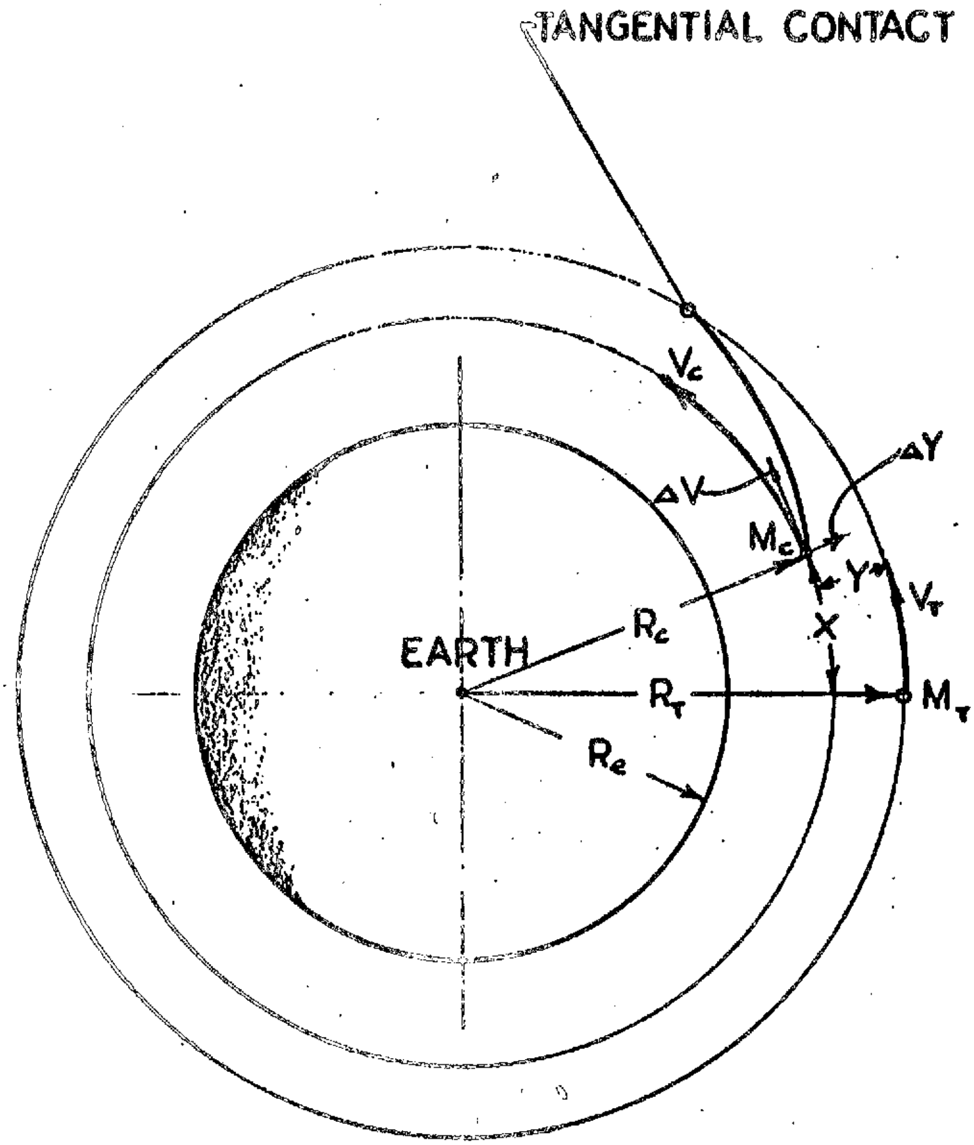
$$- 2\,\frac{\Delta\dot{Y}}{W_a}\left[1 - \frac{1}{\sqrt{\left(\frac{\Delta\dot{Y}}{2\Delta\dot{X}}\right)^2 + 1}}\right]$$

Where X = Separation distance in tangential direction
Y = Separation in radial direction
$\Delta\dot{X}$ = Change in velocity in tangential direction
$\Delta\dot{Y}$ = Change in velocity in radial direction
W_a = Orbital velocity of chaser vehicle

Equations (1) and (2) were developed assuming that the chaser vehicle in the lower orbit lags the target vehicle in the outer orbit as shown on Figure 2-15 at the instant that the orbital transfer is initiated.

When a limiting case is applied of making rendezvous in the first quadrant, it is found on solution of these equations that the chaser vehicle cannot achieve rendezvous tangentially within this restriction if it lags the target vehicle by more than .36Y. Achieving rendezvous in the first quadrant is desirable for two reasons. First, it allows the mechanical equipment used for strike, grab, and orientation to operate in the earth's shadow so that harmful thermal effects on this hardware will be minimized, and secondly it precludes the exposure of personnel to the hazards of observing attachment operations in the bright light and strong ultra-violet field of the sun.

An investigation was made, therefore, into the case where the chaser vehicle leads the target vehicle prior to injection to determine if a greater probability of making first quadrant rendezvous existed with this method than with the chaser lagging.



VEHICLE POSITION
IN ORBIT FOR RETRIEVAL

FIGURE 2-15

The equation for Y, radial separation, for this condition is the same as equation (1) in the previous case, while a change in sign is made for the X equation as follows:

$$(3) \quad X = 3\left(\frac{\Delta\dot{X}}{W_a} - \frac{Y}{2}\right) \text{Arc Tan}\left(-\frac{\Delta\dot{Y}}{2\Delta\dot{X}}\right) - \frac{4\Delta\dot{X}}{W_a}\,\frac{1}{\sqrt{1+\left(\frac{2\Delta\dot{X}}{\Delta\dot{Y}}\right)^2}}$$

$$+2\,\frac{\Delta\dot{Y}}{W_a}\left[1-\frac{1}{\sqrt{\left(\frac{\Delta\dot{Y}}{2\Delta\dot{X}}\right)^2+1}}\right]$$

Upon examination, it turns out that with the chaser vehicle leading the target vehicle within a one mile range, the limiting separation distance prior to initiation of intercept, no problem exists for making rendezvous within the first quadrant. The obvious method for initiating rendezvous is, therefore, with the chaser leading the target.

Although these equations relate the desired quantities $\Delta\dot{X}$ and $\Delta\dot{Y}$ to the given conditions of X and Y separation, the specific values of the variables, $\dot{X}$ and $\dot{Y}$, cannot be readily determined for a given separation distance X and Y. This is due to the nature of these equations which require a trial and error or an iterative process for their solution. This feature along with the discarding of higher order terms when using digital computers limit the use of computers for the final attachment process. It would require an excessive amount of equipment to bring these errors down to tolerable values for performing the close-in maneuvers.

Another limiting condition to thrusting maneuvers that can be determined using the orbital equations developed is the minimum corrections feasible with the smallest rocket thrust impulses obtainable.

According to the Bell Aero-Systems Company, the minimum reliable thrust impulse that can be obtained with the use of present day jets is in the order of .007 lb-sec. Other investigations in the state-of-the-art reveal that impulses as low as .002 lb-sec. may be obtained using Hypergolic B-1-Propellant systems. Still finer sensitivities can be predicted with the use of cold gas jets with impulse ratings as low as .0003 lb-sec. Solving for the Δ velocities that result with the use of these low thrust jets with modules having an earth weight of 20,000 to 200,000 lbs by the relation

$$(4) \quad dv = \frac{F\,dt}{M} = \frac{\text{impulse}}{\text{Mass}}$$

and plugging these results into the orbital equations, it is found that correction capabilities in the neighborhood of fractions of a foot may be obtained. However, the use of such small non-throttleable impulse rockets requires either combining a number of them with larger engines, or stacking a multitude of these smaller units and firing almost continuous pulses in order to obtain the final positioning. The errors involved with the use of many rockets with multiple impulse firings become enormous and negate the benefits derived from their theoretical capabilities. Adding the other possible sources of error listed previously, the problems associated with these terminal thrusting maneuvers are compounded.

The only possible method, therefore, to accommodate this final mating condition is the use of equipment and mechanisms that have the capability of relieving initial impact loads by reaching out to grasp the target module and bring it into final physical contact with the chaser module under tolerable conditions.

2.3.5 Search Operation

The search portion of the terminal rendezvous is considered to start at the end of the intermediate rendezvous stage. At this time the chaser and target are separated by at most, 500 feet. The search phase will be completed at approximately a 50-foot separation distance, where physical contact will take place.

2.3.5.1 Terminal Conditions at End of Search

Although the ideal situation of zero relative velocity and alignment are desired at the finish of the search phase of rendezvous, the capability of the equipment used for search has been taken into account in order to establish realistic design parameters for the mechanical retrieval equipment. The following is a listing of these terminal conditions, or initial conditions for mechanical retrieval:

(a) Range separation of Modules: less than 50 feet.
(b) Pitch misalignment: 10° total.
(c) Roll misalignment: 10° total.
(d) Difference in Orbital Inclination: 1° total.
(e) Relative Velocity: Max. of 0.5 ft/sec.
(f) Total Time for Search Operation: 10 minutes max.
(g) Modules are oriented in relation to earth for altitude control.
(h) The modules can provide their own stabilization continuously during the entire retrieval operation.

2.3.5.2 Search Equipment

The equipment required for this search operation was itemized by a letter dated 17 November 1961, from Farrand Optical Company, Inc., as follows:

(a) A 100 watt tungsten heater or lamp which is the source of radiation.

(b) A triple mirror which is used to obtain a 60 degree illuminating field whose axis is coincident with the axis of the target.

<u>Chaser Module</u>

(a) An infrared tracker.
(b) A Farrand pulsed light range finder.

These two units provide a means for delivering error signals to a computer which determines the necessary change in momentum of the chaser to reach the terminal position in the minimum time at acceptable velocities.

<u>Roll Alignment Tracker</u>

In order to achieve high accuracy in roll alignment, a separate tracking system is provided. Roll alignment is obtained independent of the axes of the modules.

Both the infrared automatic tracker and the roll alignment tracker will employ solid state devices and would have no moving parts.

<u>Common Equipment</u>

(a) Infrared Horizon Seekers

Both packages will be oriented to earth through the use of horizon seekers. The horizon seekers in both modules remain on until grab is accomplished.

(b) I. R. Tracker Lock-On and Controls

The I. R. Tracker Lock-On will initiate switchover of the operation from the radar sensors but the controls on the chaser and tracker modules will continue to operate to alter the approach velocity.

2.3.5.3 Search Procedure

A search procedure that has a good probability of success is described in the following paragraphs. This procedure is divided into three check points occurring at 500 feet, 100 feet and 60 feet separation distances at which points check signals are displayed. A transfer to the AMF rendezvous and retrieval control sensors is made upon reaching the 100 foot point. Full utilization of the retrieval homing mechanism is made at 50 feet.

(a) 500 Foot Signal

This is the initial take over point for the search operation and when it occurs the infrared equipment begins its fine tracking functions. The mechanical retrieval equipment starts to extend after cover fairings have been blown away. Checking for correct functioning of equipment is begun. The chaser vehicle sends the signal to the target module to start extension of its boom.

(b) 100 Foot Signal

At the initiation of this second signal a check for I. R. Lock-On is made and an additional automatic checkout of all equipment is again completed, this time including the determination that both booms have extended. If all equipment checks satisfactorily, the search, tracking, and flight maneuvers are transferred from the radar to the I. R. equipment. The homing sensor controls on the articulated boom of the Chaser are activated and the boom starts to adjust its position so as to strike the I. R. target on the boom of the Target module.

(c) 60 Foot Signal

When the third signal is given, the final control jets on the chaser are turned on to slow its speed to a minimum value. A maximum relative velocity of 0.5 ft/sec. must be attained in order not to exceed the capacity of the shock absorbing mechanism. The rigid booms on both the Chaser and Target should come into contact within one minute, and a transfer to the AMF mechanical retrieval system begins its operation at this point.

2.3.6 Mechanical Retrieval Techniques

As previously stated, one of the basic philosophies established for the design of attachment equipment was that it be readily adaptable to various sizes and shapes of modules. Another basic design requirement for this retrieval equipment was that it have the capability for shock absorption. It was shown in earlier reporting periods that the forces on a rigid boom extending at relatively slow constant velocities were negligible. However, impact forces at the initial strike point could be severe depending on the mass and relative speeds of the booms at initial contact.

Concepts of retrieval mechanisms and techniques were made using these basic philosophies. In addition, twelve of these concepts were compared using twenty-seven evaluation parameters.

2.3.6.1 Maximum Retrieval Distance

To minimize the computer accuracy requirements and hence its weight, it would be desirable to have as long a boom as possible within practical limitations. The practical limitations are governed by the length of modules, the complications involved in the use of telescoping mechanisms, and length effects which would cause difficulty on earth due to deflections and stresses resulting from its cantilevered weight.

Since the Rigid Boom techniques have been tentatively chosen as the method offering the best possibility of success, the maximum retrieval distance that can be tolerated is dependent upon this technique. Analysis indicates that the maximum length allowable is a function of the module length, as well as the orbital mechanics effects. Since the Rigid Boom weight is extremely small in comparison to the total module weight, module stabilization and orbital path disturbances will be negligible.

For the smallest length module presently considered for the initial manned vehicle assembly, it has been determined that a 25 foot boom extension could be accommodated. This allows a maximum retrieval distance of 50 feet and offers an adequate functional interface with the terminal rendezvous phase accomplished by flight control thrusting maneuvers. If required, greater lengths can be accommodated by using telescoping or collapsible booms or in extreme cases by allowing the booms to overhang the module ends. However, both of these methods have the following disadvantages:

(a) Decreased Reliability.

(b) Increased Development Time due to the additional collapsible or telescoping technique.

(c) Physical Interface problems due to overhang.

2.3.6.2 Mechanical Retrieval Concepts

Some of the mechanical retrieval concepts that were generated as a result of this study are described in this section. Most of the remaining concepts that were evaluated have been described and illustrated in prior reporting periods.

The operations required of all the concepts are strike, grab, and initial orientation.

(A) Rigid Arm-Side Location Concept.

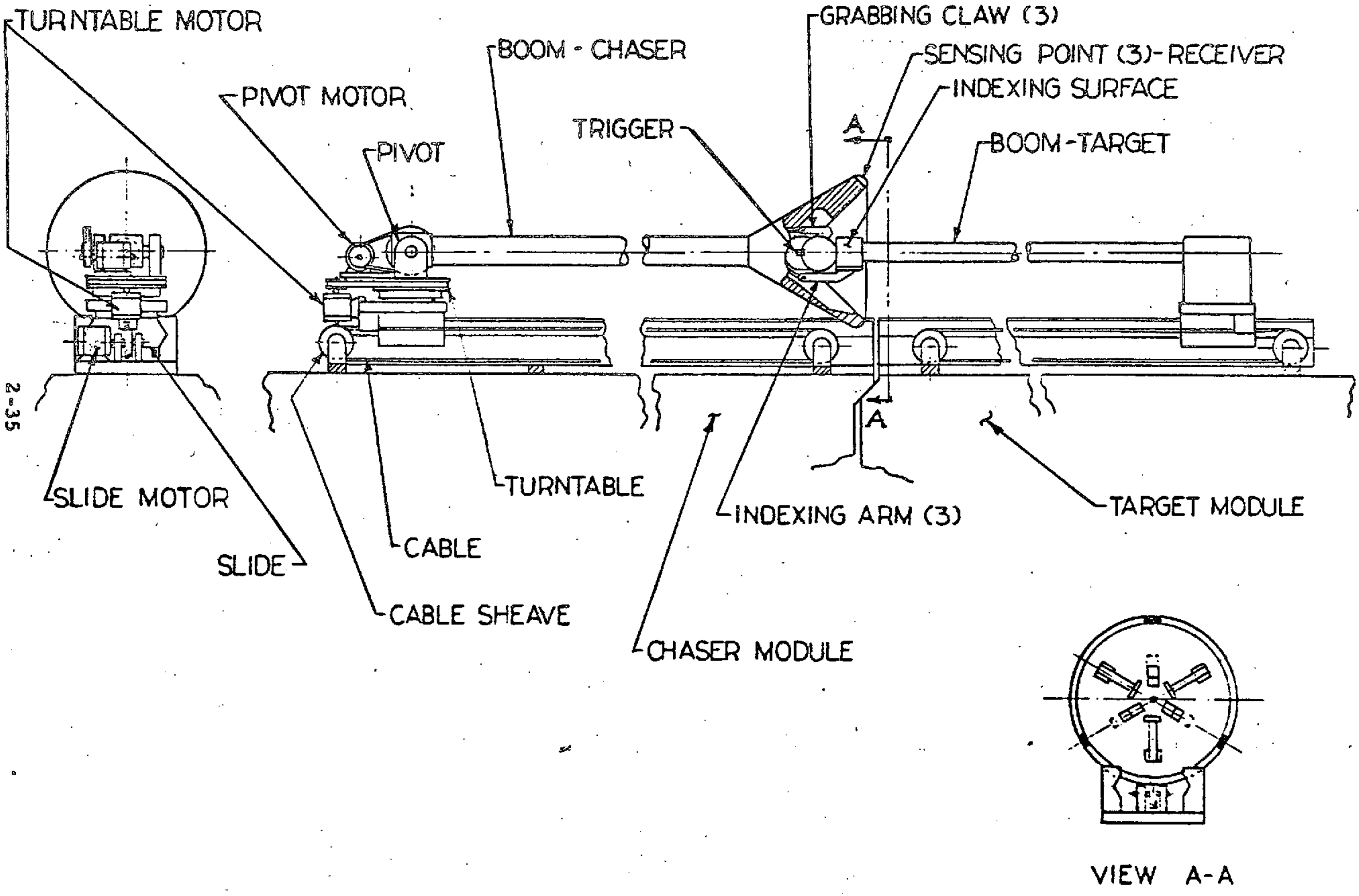
This technique is considered to offer the most promise for achieving success in performing the strike, grab, and initial orientation functions required during the final 50 feet of retrieval.

(1) General Description

This concept, shown in Figures 2-1, 2-16, and 2-17, consists essentially of extendable rigid booms mounted on the exterior surface of both the Chaser and Target modules. The chaser boom contains a conically shaped female receiving end, while the target boom has a spherical male end (see Figure 2-16). For a module which is either to be the first or last section of a completed vehicle, only a chaser boom or a target boom is present. However, for a module which is to be an intermediate section of a completed vehicle, both ends of the module require a boom mechanism.

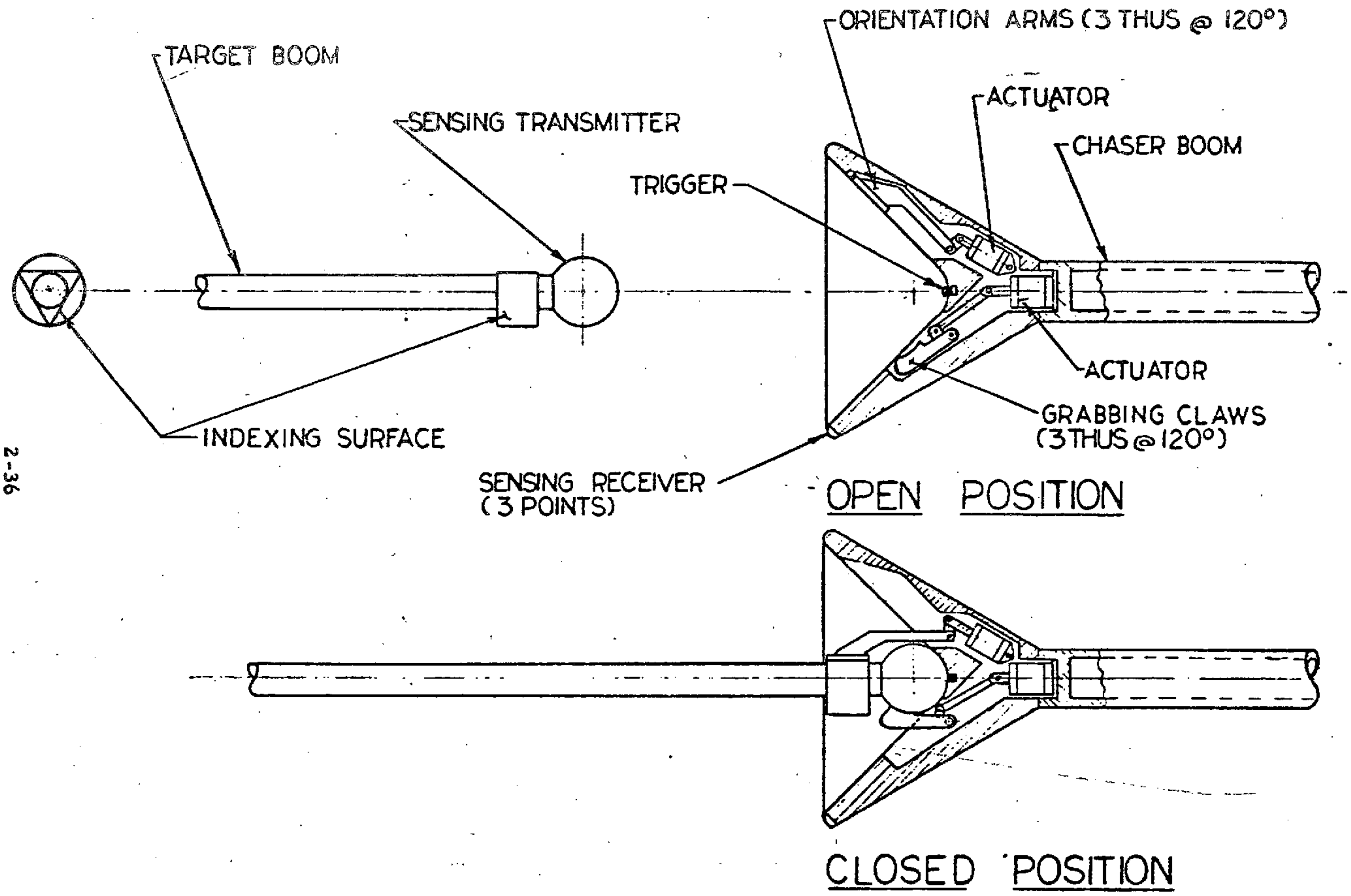
The target boom sends out infrared modulated pulses. Sensing receivers mounted on the chaser pick up these impulses from the spherical ended target transmitter (see Figure 2-17). These impulses provide the intelligence for the chaser boom positioning mechanisms to make corrections so that the target boom can more readily find the apex of the chaser conical surface. Upon making contact with the apex, a trigger mechanism on the chaser is actuated causing an explosive thruster to initiate action of three seizing claws. These claws mounted in the housing of the conical surface close on the ball ended target boom completing the strike operation by the formation of a swivel joint.

Initial orientation is started immediately following the seizure of the target boom. This action is started by the movement of three additional orientation holding arms in the conical housing. These arms apply pressure to three, three dimensional cams on the target boom surface. This pressure causes rotation of the target boom which in turn rotates the target



RIGID BOOM - SIDE LOCATION

FIGURE 2-16



GRAB & INITIAL ORIENTATION MECHANISM

FIGURE 2-17

module about its longitudinal, lateral and vertical axes to provide initial orientation in roll, pitch and yaw. The initial alignment is aided by a final operation which is the retraction of the two booms forcing the modules into closer contact and more precise alignment. A block diagram of the Retrieval operation is shown in Figure 2-18. A generalized concept drawing of the repeated retrieval process is shown in Figure 2-19.

(2) Chaser Equipment

The chaser mechanism consists of the following major components.

(1) An extendable boom.

(2) Extension, rotation, elevation and shock absorption mechanisms.

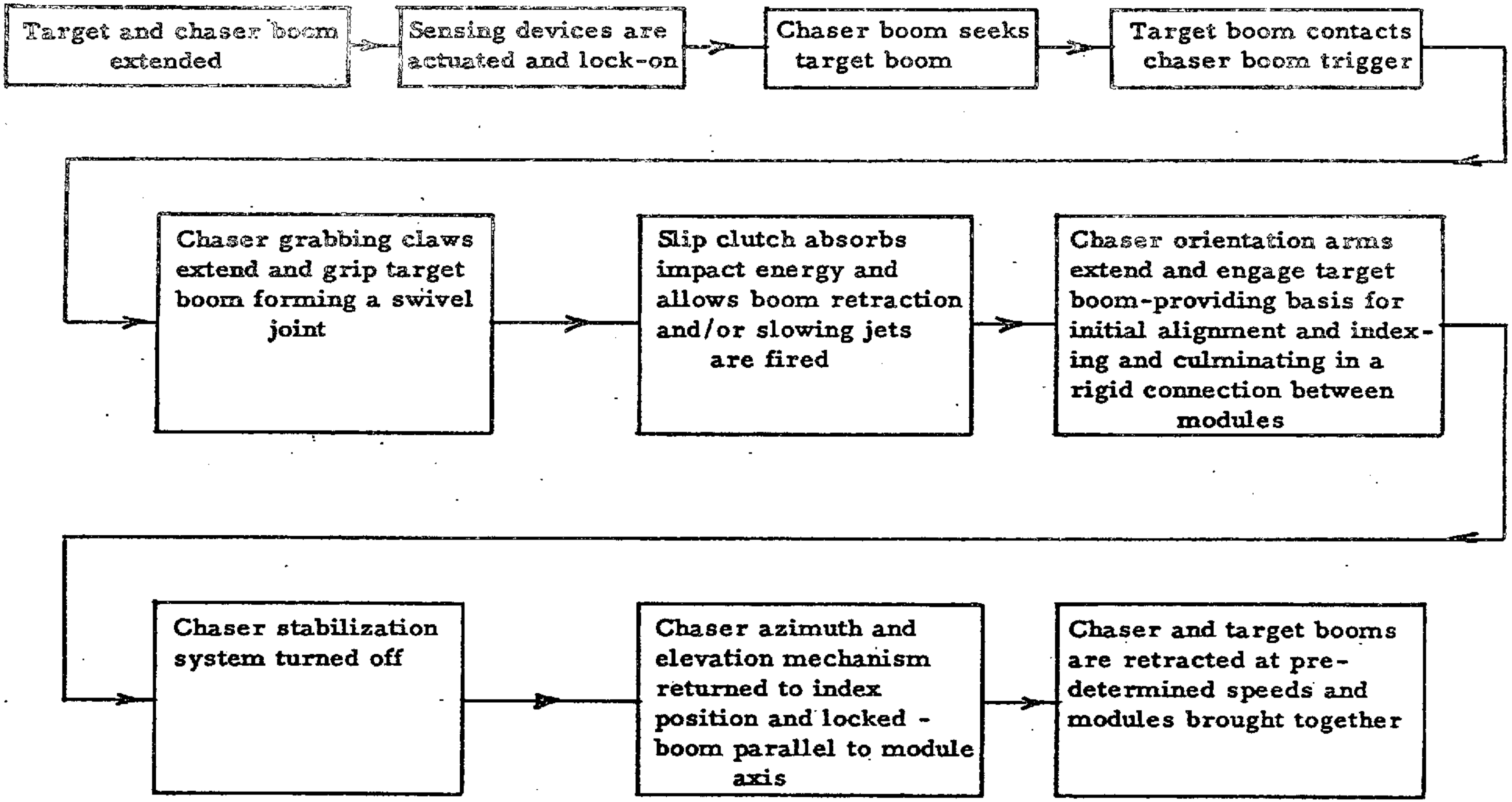
(3) Grabbing and Holding mechanism

(4) Sensing Receivers.

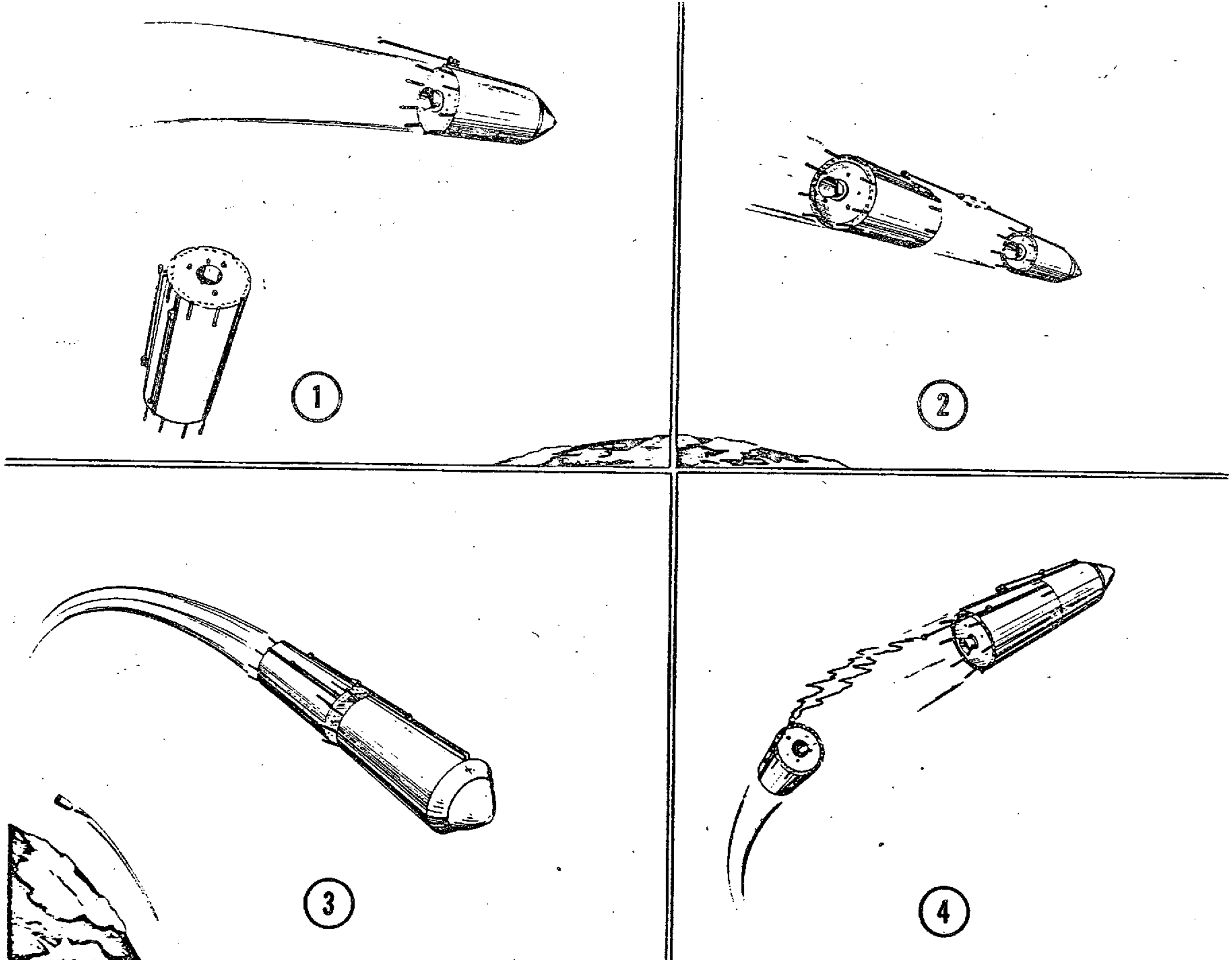
The extendable boom is a lightweight tube, attached and pivoted at one end on a rotating turntable. It contains the grabbing, holding, and sensor equipment in the conical end.

The extension mechanism consists of a slide attached to an endless cable running over two pulleys which are in turn driven by a electric motor. The slide moves in V-shaped guide rails carrying the turntable and boom along with it. An alternate means of providing the extension could be a chain and sprocket arrangement in lieu of the cable and pulleys. As the normal force on the guide rails is negligible due the weightlessness of objects in orbit, sliding friction will not be a major consideration. However, in order to prevent possible seizure due to friction welding in a vacuum, the rails, pulleys, and other friction surfaces will be teflon coated. The normal forces produced by the coriolis effects will also be of minor consequence because of the slow speeds at which the extension and retraction are conducted. The impact forces causing column action in the boom, however, could be severe. To alleviate this type of loading, the pulley or sprocket drives will contain a ratchet type coupling which allows the slide to be driven in either direction from the motor end but disengages when the conically shaped end of the boom creates the driving force. When disengagement of the motor drive occurs, a brake mechanism is actuated which provides the resistance to slow down the reversed motion caused by too high an initial impact.

The rotational mechanism consists of a turntable driven by a motor so as to provide azimuth control for the boom. Another motor provides elevation control of the boom. The boom in this instance pivots about the



BLOCK DIAGRAM OF RETRIEVAL OPERATION
(RIGID BOOM - SIDE LOCATION)
FIGURE 2-18



GENERALIZED RETRIEVAL PROCESS

FIGURE 2-19

turntable base. It is contemplated that small lightweight servo motors will provide the power required to control the chaser boom motions. The boom sensors will provide the error signals to position a rate damped servo system, while the servo motors will make the corrections required.

The grabbing and holding mechanisms will operate as described previously. The electrical wiring required for their operation will be housed within the boom tube.

(3) Target Equipment

The target equipment employed in this concept contains a boom which is only capable of extending and retracting. It does not contain azimuth or elevation mechanisms. The ball ended portion of the boom contains the sensor transmitter.

(4) Concept Advantages

(a) This concept can be easily applied to any module situation with only minor changes in the mountings.
(b) The location of the mechanism on the exterior module surface offers no interference to module mating interfaces.
(c) This concept can be adapted to manual control with visual capabilities as there is no restriction of view from the interior of the module.
(d) The exterior location provides for easy ejection of mechanisms if desired after assembly.
(e) This concept uses simple reliable components that are considered off-the-shelf items for earth use.

(5) Concept Disadvantages

(a) The exterior location requires special fairings for streamlining during earth launch.
(b) All components must be compatible with space environments for the entire operation from earth launch to completion of retrieval, otherwise all components must be hermetically or environmentally protected or sealed while in the stored position.
(c) For adaptation to different mating module sizes and shapes, the location of components around the module periphery requires extended struts which increase drag at launch and would produce large bending moments at the module surface at impact. Neither of these disadvantages is serious from a structural consideration.
(d) Friction drives have inherent slip characteristics which reduce the accuracy of servo system.

(B) Rigid Boom — Center Location Concept

This concept, shown in Figure 2-20 is identical in principal to the operation of the Rigid Boom — Side Location Concept previously discussed. The mechanisms required however, are somewhat simplified by the nature of the position, but the concept pays for this advantage in a higher weight structural requirement.

Other advantages of this concept are: that the center position provides a better condition for alignment and indexing; and the outside shape of the module has little affect on the supporting structure. The greatest disadvantage of this concept is that it requires valuable space inside of the module for storage both before and after retrieval. Sufficient clearances would have to be provided to avoid interference with the operation of other components or subsystems in the same portion of the module.

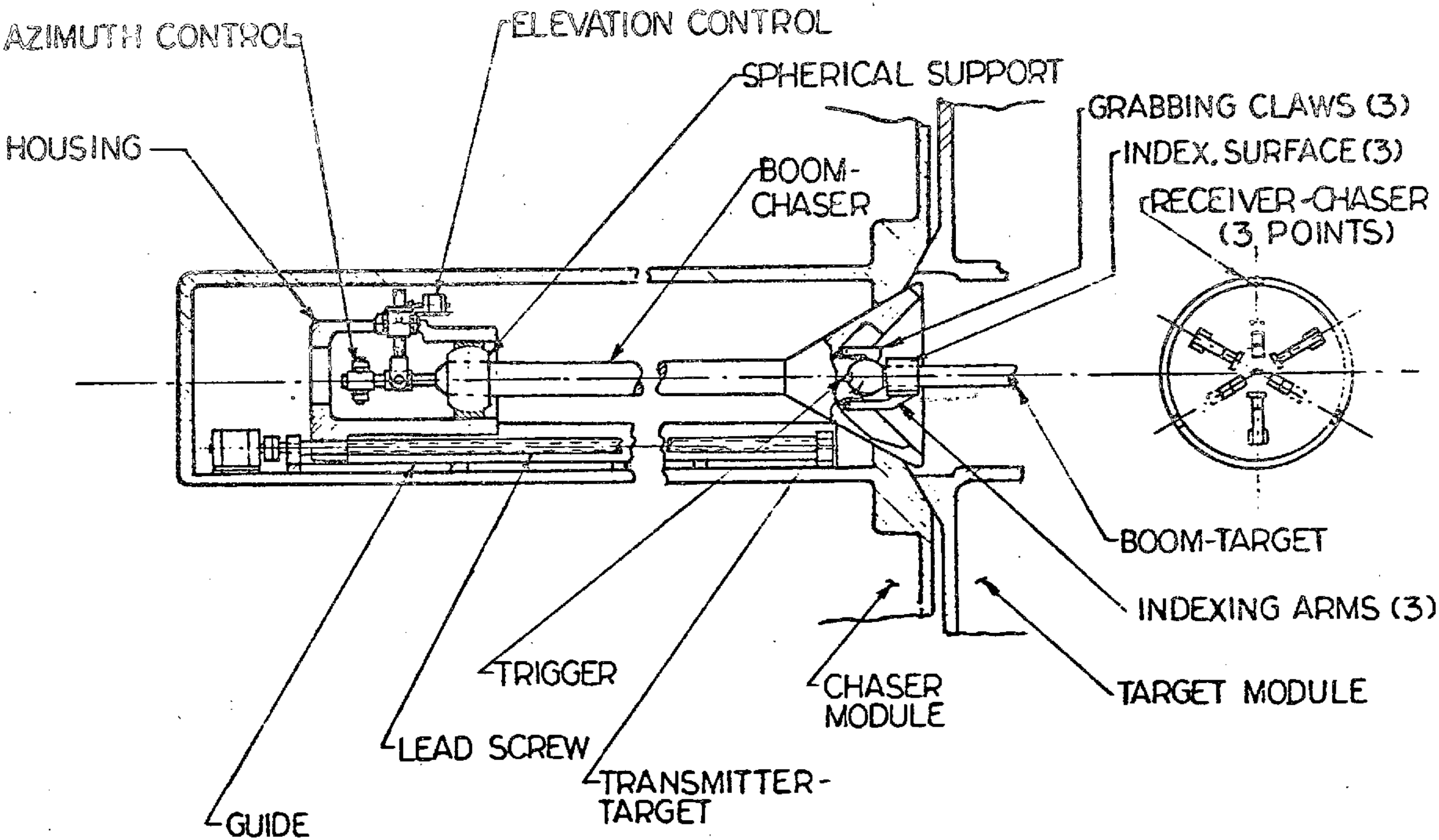
(c) Passive Dart Concept

In this concept, shown in Figure 2-21, a dart is aimed at a target and when it is centered and at the proper distance, it is ejected toward the target. The target is a cryogenic magnet having a magnetic flux density of over 50,000 gauss, and weights only about two pounds. The dart is attached to a cable which is payed out by a mechanism. After the dart attaches itself to the target, the mechanism retrieves the dart and the target module simultaneously.

The concept is exceedingly attractive since it is simple, light in weight, and quite compact. It is now receiving a good deal of attention in our in-house space programs and will be reported on in detail at a later date.

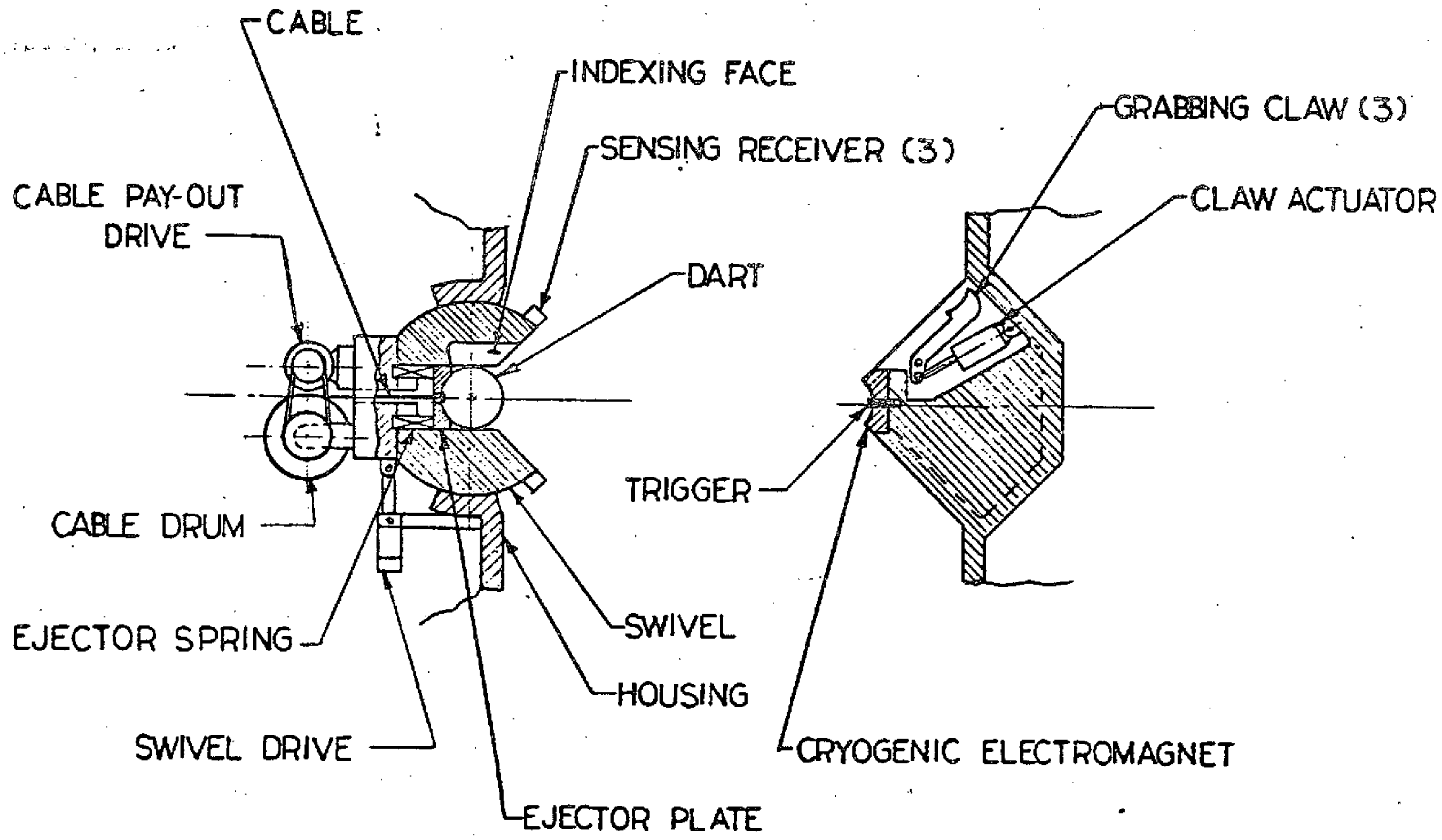
2.3.7 Evaluation of Mechanical Retrieval Techniques

A total of 16 concepts for retrieval were evaluated during this reporting period. The dozen best of this group were carefully evaluated against twenty-seven factors for various possible configurations. As would be expected, the superiority of one technique over another depends on the particular configuration for the operation and the dominant factors for rating the tasks involved in performing the operation. However, the Rigid Boom — Side Location concept was consistently rated best. The results of one set of evaluations is shown in Figure 2-22 together with a list of the twenty-seven evaluation factors.



RIGID BOOM-CENTER LOCATION

FIGURE 2-20



PASSIVE DART

FIGURE 2-21

RETRIEVAL CONCEPT EVALUATION FACTORS	
1.	Performance risk
2.	Complexity
3.	Ease to adapt to various geometrical forms of modules (Flexibility)
4.	Development risk
5.	Foolproofness of operation
6.	Development time and requirements
7.	Safety to the structure in case of malfunction (and to crew if manned)
8.	Testing time and costs
9.	Ability to detect malfunction
10.	Restriction to fastening technique
11.	Reliability with respect to mission success probability
12.	Ability of exact alignment and indexing
13.	Reversability of functions in case of malfunction
14.	Weight of system
15.	Mechanism storage space required in module: a. Prior to retrieving operation b. After completion of joining operation
16.	Tolerance requirements
17.	Use of special materials
18.	Impact control
19.	Pull-up speed control
20.	Time to effect operation
21.	Ease of maintenance and repair
22.	Thermo expansion and contraction effects
23.	Human factors requirements
24.	Power requirement
25.	Cost of development
26.	Cost of fabrication
27.	Fabrication ease

EVALUATION

CONCEPT		TOTAL RATING
1.	Rigid Boom - Side Location	1801
2.	Rigid Boom - Center Location	1721
3.	Passive Dart	1685
4.	Screw Jack Retrieval	1671
5.	Foldable Arms	1460
6.	Inflatable - Cable Retrieval - Manual	1456
7.	Active Dart	1389
8.	Active Dart - Scanning	1359
9.	Active Dart - Harpooning	1359
10.	Active Dart - Cable Retrieval	1312
11.	Inflatable Arm	1304
12.	Rigid Telescopic Arm - Inflatable Target	1248

SUMMATION OF RETRIEVING CONCEPTS
EVALUATION
FIGURE 2-22

The figure shows that the technique ratings fall into three groups with four concepts in the top group, two in the middle group, and six in the lowest group. All of the techniques discussed in this section were among those in the top rated group. It is important to note that assignment of different sets of weight factors appreciably changes the placement results shown in the tabulation. To realistically prove the feasibility or superiority of any of these techniques requires more information than is presently available. A development program is clearly indicated at this point with testing both on earth, in simulators, and in orbit. Only then could conclusive results be established. It appears from our study thus far, that the most promising technique with which to start such a development program would be the Rigid Boom - Side Location concept.

2.4 DOCKING AND ASSEMBLY

2.4.1 Introduction

The docking and assembly requirements which were investigated during this phase of the OLO program were directed exclusively toward the early accomplishment of the manned lunar landing and return mission. Figure 2-3 contains an operation and task breakdown for this portion of the mission. In the docking and assembly area these tasks are; pull-up, alignment, indexing, mating, interface connections, sealing, unifying, crew transfer, propellant transfer, and checkout. The integration of these individual tasks into a docking and assembly system combined with the necessary hardware concepts, has been aimed at achieving a reliable and flexible system with a minimum projection of the state of the art. Optimization, sophistication, and the utilization of more advanced techniques has been sacrified in favor of increasing projected reliability of operation, and reducing development time and development risk. Later on, programs can be established which are based on the advanced knowledge of the detailed space experience obtained in the interim. Since these will be essentially free from the present pressures of the space race they can provide the more optimized and sophisticated system.

The criteria listed below was developed from a combination of the factors learned in the initial OLO program, in current in-house studies, and in the early phase of the present extension program. They found considerable use as a guide during the origination of hardware concepts as well as during evaluation. Other criteria pertinent to the conceptual design is presented in the General Consideration section of this report. Criteria variations and modifications applicable to specific concepts are presented in the detailed concept discussion.

General Criteria

1. Relative position of modules at initiation of the docking and assembly operation is less than 8 inches.

2. Relative logitudinal misalignment at start of operation = $\pm 1.0^{\circ}$.

3. Relative indexing misalignment at start of operation = $\pm 1.0^{\circ}$.

4. Relative speed of modules at start of operation = 0.1 fps max.

5. Reaction time for docking and assembly operations will be less than 10 minutes.

6. All concepts will be investigated to determine the need for devices to indicate satisfactory completion of the critical tasks such as interface connections, sealing, and propellant

transfer. Indicators will be incorporated as an integral part of the mechanisms wherever it is deemed feasible.

7. The ability to correct malfunctions shall be incorporated in the system where necessary to insure feasibility of performing the function.

8. The simplest and most "fool proof" mechanisms are desired.

9. All components and parts required for the assembly must be reliably attached to the face of one of the modules being connected. (No loose parts or individual emplacements will be allowed.)

10. All connections and fasteners must have positive locking.

11. Structural integrity shall be provided to accomodate bending, axial, and torsional stresses developed in connectors and fasteners.

12. Interface fasteners shall be compatible with normal module splice designs.

2.4.2 Common System Component Tasks

Pull-up, alignment, indexing, mating, interface connections, sealing, unifying, and checkout are the major tasks which are common to all concepts under consideration. Satisfactory solution of these tasks provides the capability to form a basic sub-system. Each of these areas was investigated to establish guide lines and background information for the conceptual design phase.

2.4.2.1 Alignment and Indexing

The alignment and indexing required for the mating and assembly of the OLV modules has been divided into three progressive steps. The initial gross alignment and indexing is provided by the stabilization and attitude control system just prior to the start of the retrieval operation. During the retrieval operation the second orientation correction will be applied. Relative misalignments of $\pm 1^{\circ}$ for indexing and $\pm 1^{\circ}$ for longitudinal axis alignment are specified as the final condition for this step. Mechanism concepts for the docking operation will provide the final indexing and alignment correction. The methods proposed for this final alignment and indexing include the use of one or a combination of the following; gross conical lead-in, conical lead-in by pin mating, cam surfaces, external guides, and wedges. The selection of the method best suited to perform the task is dependent on related tasks required of the docking mechanism. Two examples of dependent related tasks are: the pull-up mechanism which offers the best method of supplying the forces required to actuate the alignment-indexing mechanisms; and the mating requirements which are a function of the type and complexity of interface connections. A positive conclusion reached during this preliminary analysis was the decision to eliminate axial alignment and rotational indexing errors by a series of progressive steps, thereby providing a gradual closing-in on the required tolerances.

2.4.2.2 Pull-up and Mating

Pull-up is the mechanical process by which final closure of any two modules is made. It is divided into two phases; the initial phase being provided by the retrieval mechanism which brings the modules within 8 inches of each other, and the final phase which is provided by the docking mechanism. The final pull-up phase supplies the forces required to complete module interconnections and insure positive seal pressure, thereby mating the modules to each other. The necessity for a mechanical pull-up derives from a preliminary error analysis performed during the orbital flight study discussed in the retrieval section of this report.

2.4.2.3 Interface Connections

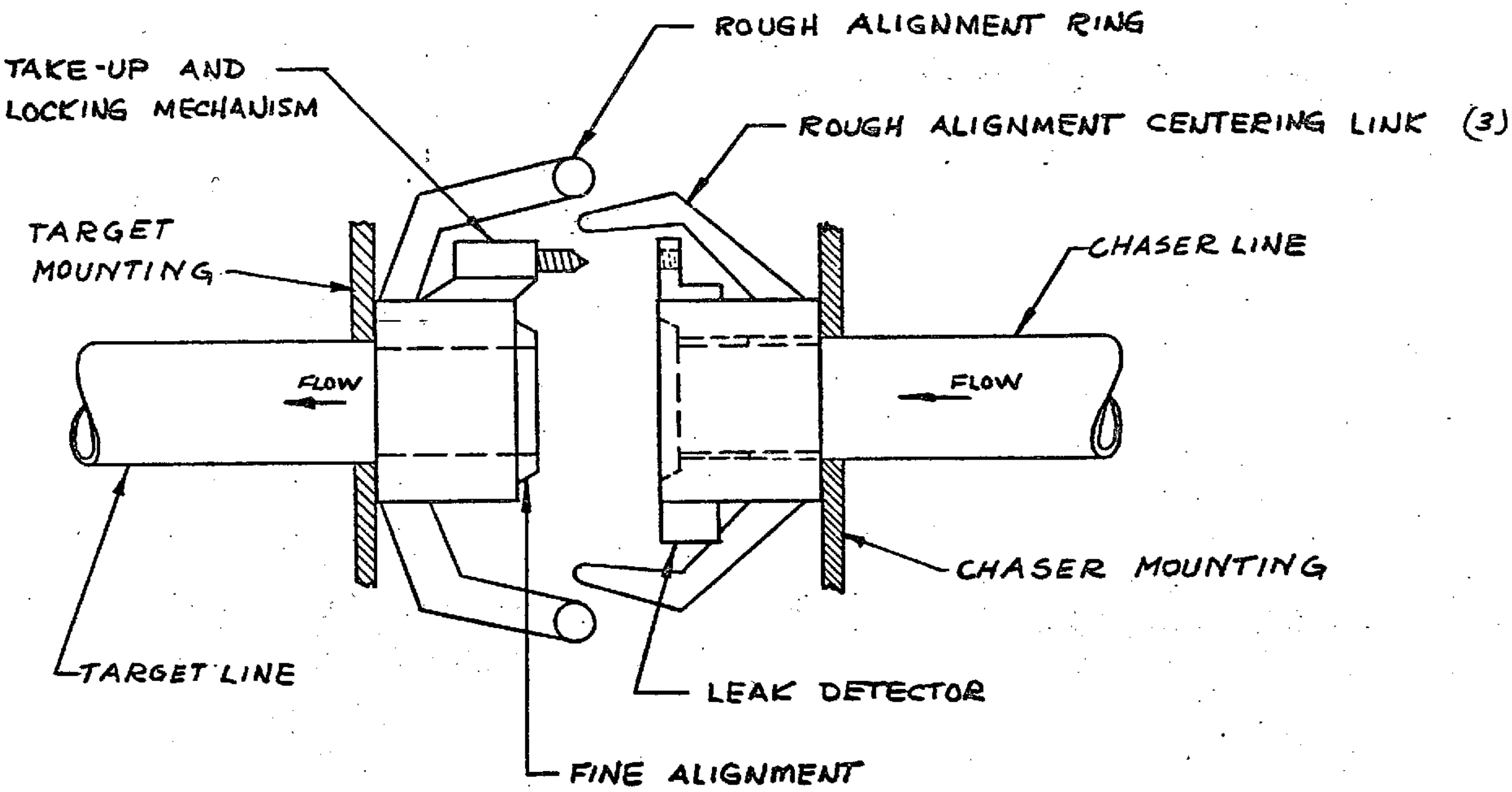
A separate interface survey report has been made in accordance with Marshall Space Flight Center requests. The results of this study, which collected data on Ranger-Agena, Mercury-Redstone, Mercury-Atlas, Mariner R-Agena, Mariner A-Centaur, Surveyor-Centaur, Saturn SI-SIV, Thor-Agena, Thor-Abel, Delta, Transit and Discoverer, indicate the feasibility of using connectors similar to the cryogenic type quick connect and disconnects presently in use, as shown in Figure 2-23. The connectors will be mounted one half to each of the two mating faces. The female connector will utilize a floating cartridge mount, thereby allowing for the final tolerance take-up. Additional alignment and orientation correction will be accomplished through the use of conical lead-in holes and tapered or round-ended locating pins. These pins will be made in varying lengths to eliminate the probability of jamming. It is presently expected that individual take-ups now incorporated on current commercial connectors will be required to insure positive engagement of the mating parts. Examples of available hardware requiring a minimum amount of rework to meet the requirements of an automatic orbital engagement are:

"Cannon" or "Wiggins" type multi-pin seal type connectors for electrical interconnections.
"On-Mark", "Wiggins", or "Snap-Tite" fluid transfer quick-connect and disconnect type couplings similar to those designed for Titan cryogenic propellant transfer.

2.4.2.4 Sealing and Test

(a) Study Conclusions

Analysis of various orbital hardware concepts to determine the best methods of solving sealing problems indicates the possible use of many



CRYOGENIC COUPLING

FIGURE 2-23

varied concepts. Most accepted earth methods with some modification and many of the unusual schemes proposed in the original Orbital Launch Operations study report find use in one hardware concept or another. These conclusions point to the definite need for many proven concepts thereby providing the space hardware designer with a manual from which he can pattern the seals for his specific hardware. This manual should categorize and rate a wide variety of seals. Its initial publication should be made at the earliest possible date thereby providing the hardware designers with a tool to accelerate their development programs. It should be arranged to allow for continual expansion both in the addition of new categories and an increase of the number of proven concepts within any one category.

In addition to the basic problems of developing and verifying suitable materials to withstand the effects of the orbital and lunar environments, much work must be done in the area of leak detection apparatus, inspection techniques, and seal maintenance or replacement, where reliable long life seals cannot be developed. Since manned systems are the basic consideration, all seals whose failure or excessive leakage may prove detrimental to crew safety should be provided with leak detection equipment and a positive crew warning system. The hardware and seal designs must provide for sufficient crew reaction time or automatic means to either repair seals, close down faulty systems, or otherwise prevent possible catastrophies.

Keeping in mind our prime mission of an orbital launch to the moon in the early part of 1967, the following general categories of seals were chosen:

a) Fluid and Electrical Couplings
b) Mechanisms and Mechanical Extensions
c) Air Lock Doors
d) Repair Seals

For best utilization in a design manual, it is recommended that these categories be further subdivided and serve as the basis for the initial manual publication. The recommended categories will first be divided into static and dynamic groups and then be enumerated in relation to a specific function. Within these functional categories, additional subdivisions should be made according to physical and environmental use. Fluid couplings, for instance, can be categorized in relation to the major properties of the fluid medium being served, therefore leading to possible subdivisions such as cryogenic seals, high pressure pneumatic seals, hot gas seals, etc. Physical breakdowns would be based on size and configuration. The Table of Functional Seal categories, which follows, comprises the recommended initial breakdown.

TABLE OF FUNCTIONAL SEAL CATEGORIES

Static Seals	Dynamic Seals
1. Individual Fluid Pipe Connections	1. Reciprocating Shaft Extentions
2. Individual Electrical Wire Connections	2. Rotating Shaft Extensions
3. Multi-Fluid Pipe & Electrical Wire Connect Panels	3. Air-Lock Doors
4. Repair Seals	

The items listed represent the major functional categories. Subdivisions to account for physical and environmental differences have not been shown. They will be best developed in the primary phase of the recommended seal program.

(b) Basic Criteria

The following General Criteria and Criteria for Evaluation were developed during the initial seal investigation and are presented as a guide to the scope and relative importance of the basic seal parameters and requirements.

General Criteria

The seals considered in this study must function in earth, lunar, and orbital environments. Specific restraints and constraints caused by these environments and other operating conditions are:

1. Hard Vacuum - About 10^{-9} mm Hg.
2. Exterior Temperature - +250°F to -250°F.
3. Useable Life - Varies dependent on system use requirements, replacability, and use of external protective means. Since initial manned lunar flights and short duration stays are the main consideration of this report, a minimum life of 6 months for seal systems will be considered satisfactory.
4. Positive indication of seal integrity must be provided or inherent in all seal concepts.

5. Types of seals to be considered are those having specific use during orbital assembly operations. The following list comprises all types which will be considered by this report.

   a) Fluid and Electrical Couplings
   b) Mechanisms and Mechanical Extensions
   c) Air Lock Doors
   d) Repair Seals

6. Size of Seals - All sizes from extremely small connections to 35 foot diameter modules will be considered. In general, it is felt that the size range from small holes to Air Lock Doors will comprise the bulk of seal problems for orbital assembly.

7. Seal Performance Time

   a) For space assembly work, most seals will be preinstalled integral with the modules. Seal activation will occur automatically during the mating and interconnection phase. Some seal concepts (i.e., Inflatable gaskets; resin adhesive, and potting) may be completed subsequent to mating. No specific time limits will be set for these concepts but evaluating criteria will take cognizance of time durations.

   b) For repair seals, minimum times must be assessed in relation to system requirements of permissable down time, permanent damage due to leakage, and crew safety.

Criteria for Evaluation

Listed in General Order of Weighted Importance.

1. Seal effectiveness (performance)

2. Seal reliability

3. Durability (life of seal)

4. Ease in incorporating leak detection indicators

5. Development risk (confidence level)

6. Development time

7. Ability to correct for inexact alignments

8. Ability to operate under varying tolerance

9. Material availability (state of the art)

10. Maintainability (by replacement or repair or self correction)

11. Weight of system

12. Ease of development testing

13. Reusability

14. Complexity

15. Speed of sealing

16. Power requirements

17. Ease of fabrication

18. Development cost

19. Effect on mating structures

Rating criteria and factor weights must be assigned based on the specific application and its importance to the mission. Reliability and time are the present emphasized goals, optimization and sophistication will come later.

2.4.2.5 Unifying

Unifying is the permanent joining of the mated modules. It is the addition of a positive assembly device which furnishes a redundant hold to the fastening provided by the mechanical coupling mechanism. Techniques to accomplish this task are well within the projected state of our technology, and have been adequately analyzed and evaluated in the assembly section of the initial OLO contract report. Representative concepts of the simpler and more reliable methods for automatic operation include explosive pins or spikes, pyrotechnic forming, and snap rings.

Since separation of the modules is often required by the basic space-vehicle design; either a means of breaking the fastening device, or a different separation point is required. Regardless of which of these methods is chosen, present applications dictate the use of pyrotechnic separating devices, primarily explosive bolts. The Interface Survey Report recommended the use of a single assembly and separation plane in addition to pyrotechnic separation. It is therefore recommended that a pyrotechnic means of breaking the fastening device be incorporated in its design.

2.4.2.6 Checkout

Checkout provides the transition between assembly and launch. It involves the necessary inspection to insure the proper operation and condition of space-vehicle equipment and commodities, with particular emphasis on those items which had an interface connection. In conformance with the statement of work for this portion of the program, no additional detailed work has been performed by AMF in this area during this reporting period.

2.4.3 Docking and Assembly Systems and Evaluation

2.4.3.1 Summary

The system concept and evaluation phase of the docking and assembly operations was divided into four steps, as follows:

1) Review of existing concepts
2) Projection of new concepts
3) Preliminary evaluation to choose best group of concepts
4) Final evaluation to choose best concept.

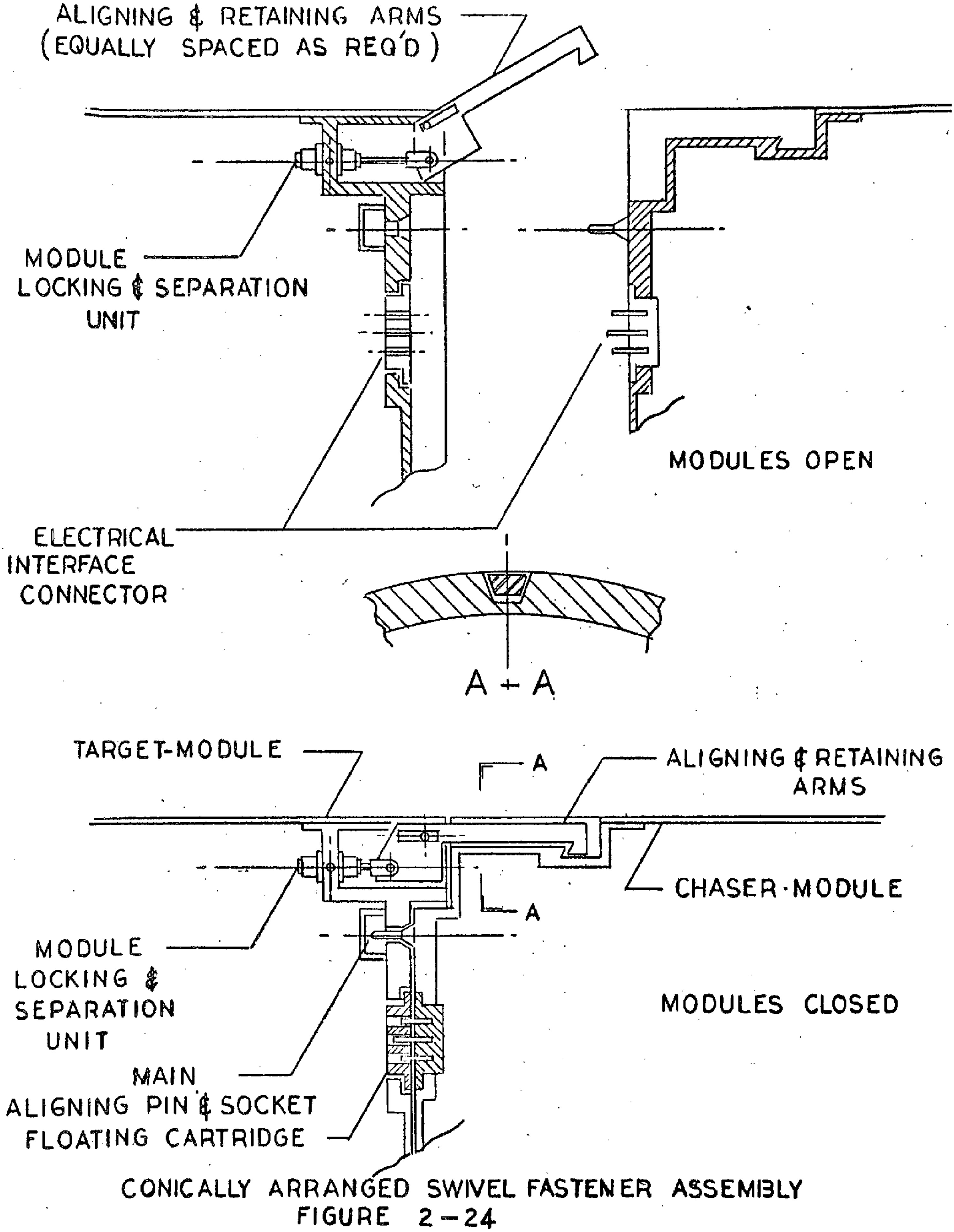
Guidelines and background information developed for the component tasks of the docking and assembly operations, and general criteria derived from these task studies, served as the basis for the performance of steps one, two, and three. Work in these areas culminated in the choice of four concepts. These concepts are: the Conically Arranged Swivel Fastener Assembly; the Snap Wedge Assembly; the Key Pin Assembly; and the Bell Crank Assembly. These four concepts were then further analyzed with respect to the detailed criteria established for the final evaluation. The evaluation was then made and resulted in the choice of the Conically Arranged Swivel Fastener Concept.

This concept is illustrated in Figure 2-24 and a block diagram of its operations is shown in Figure 2-25.

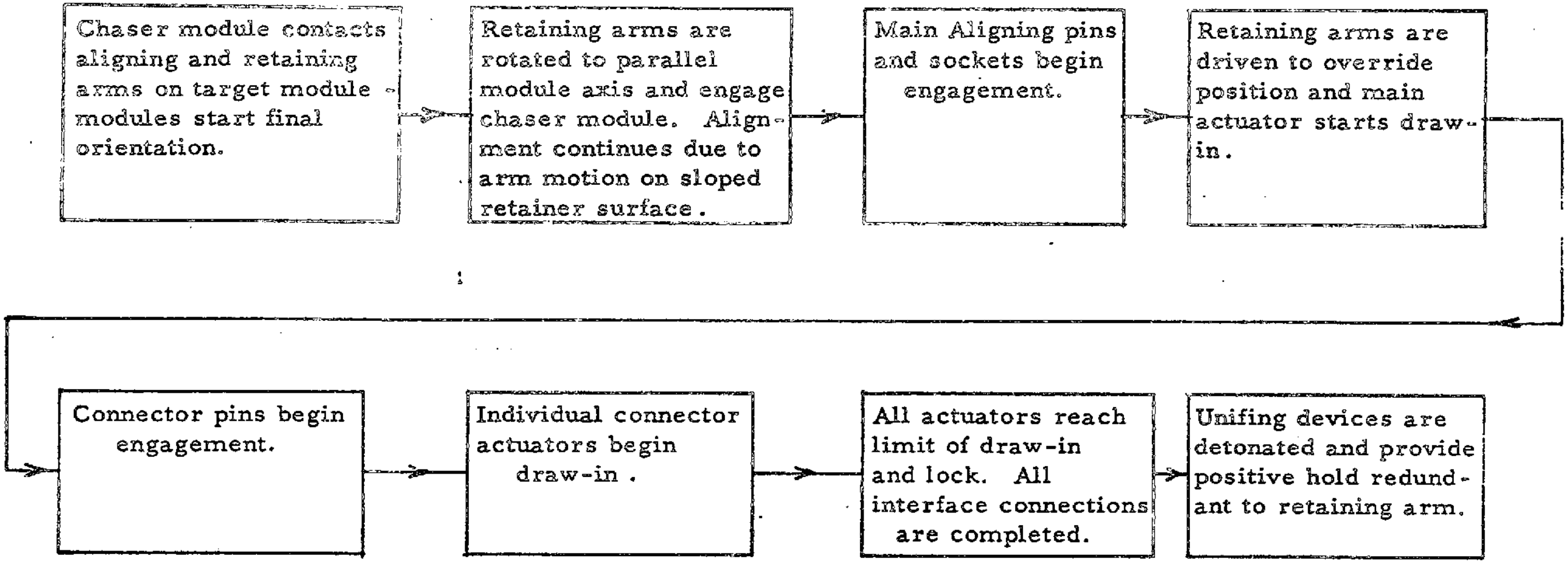
2.4.3.2 Conically Arranged Swivel Fasteners

(a) General Description

The conically arranged swivel fasteners concept, Figure 2-24, is composed of a number of identical sub-systems equally spaced around the module peripherey. The exact number of fasteners required is a function of the module diameter, stress transfer levels at the interface plane for the various loading conditions, and the system interface connector requirements and arrangement. Cursory analysis of existing space vehicle interfaces indicates that eight (8) sub-assemblies placed at 45° may be sufficient. If more stress transfer points or special hold locations are required, separate unifying devices (probably pyrotechnic) could be supplied.



CONICALLY ARRANGED SWIVEL FASTENER ASSEMBLY
FIGURE 2-24



BLOCK DIAGRAM OF DOCKING AND ASSEMBLY OPERATIONS
(CONICALLY ARRANGED SWIVEL FASTENERS CONCEPT)
FIGURE 2-25

The main components of each fastener sub-assembly are:

1. Aligning and retaining arm - this arm is about two feet long, tapered in cross-section, with a hook lip at the extended end and a bell crank lip at the mounted end. It is attached to the target module.

2. Module locking and separating unit - this unit provides the control force necessary for the final pull-up and closure pressures required between modules. It is trunion-mounted at one end to a rigid portion of the module, and pin connected to the retaining arm bell crank lip.

3. Retaining arm wedge groove - this groove is provided on the exterior of the chaser module. Its sides are sloped to allow module indexing when engaged by the retaining arm, and a recess is formed at its end, thereby providing the retaining arm hook lip with a positive engagement and pull-up surface.

4. Main locating pin and socket - this unit provides the final alignment and indexing error correction for the module. The pin has a tapered or rounded end and is projected from the chaser. The socket, located on the target module, is designed with a conical lead-in for ease in pin engagement.

Since the duration of storage in a parking orbit may be appreciable it is recommended that light weight shrouds or fairings be placed over exposed mechanisms and connectors to prevent damage by the hostile environment. Provisions should be made to dissengage and blow away these shrouds just prior to assembly.

(b) Discription of Operations

The first function performed by this concept is the initial guidance of the chaser module to the target module through a relatively wide conical configuration inlet. This configuration is formed by individual arms located on the periphery of the module. Upon initial contact, the arms guide the chaser module into a close relative longitudinal alignment and rotational indexing with the target module. As the modules move closer together, the chaser contacts the bell crank lip which is an extension from the retaining arm, and causes the arm to rotate to a direction parallel to the module axis. As the rotation progresses the arm, which has a tapered cross-section, engages wedge grooves similar to splineways on the outer surface of the chaser module. This action provides an additional step in the progressive elimination of the alignment and indexing errors and culminates in the engagement of the target arm hook lip with a recess at the end of the chaser wedge groove.

Continued closure between the modules allows the main alignment pins to enter the conical lead-in sockets and drives the retaining arm actuating cylinder to the end of its overide position. At this point the closure control force is transfered from the retrieval mechanism (Rigid Boom - Side Location) to the retaining arm actuator. The final orientation correction of the module structures has now been completed.

As pull-up is continued by the retaining arm actuator, interface connectors will engage in sequential order. Pending the exact nature of these connectors, individual actuators will be provided where required to insure positive engagement. These actuators will be activated as mating connectors are contacted. The requirement for fine alignment and indexing between connector mates will be accomodated by movement of the female connector within its floating cartridge connection.

The next and final step in the sequence of operation is the engagement of the unifying devices. For this concept initial studies and shop tests are recommended to determine the feasibility of utilizing the main aligning pins as the unifying device.. Two pyrotechnic charges will be placed within the pin; one at the lead end, and one at the center. The end charge will be detonated at final closure and will create a flaring fastening and tightening similar to a rivet connection. The second charge will be implanted across a necked-down or pre-weakened plane and will be used for later separation. A closed retainer shield will be pre-mounted around the pin end to prevent damage to module components from exploding debris during separation. If the mating pin concept does not prove feasible, separate explosive pins or spikes would be provided to perform the task. These separate devices would be provided with a pyrotechnic means of disassembling when separation is required. Separate pins or spikes may also be provided if mating pin concept proves feasible to increase the number of hold points or provide special hold points between fasteners.

The exact forms of seals have not been specified, but it is expected that all connectors will be provided with some type of seal. Normally, face seals are better for this general type of connection. These seals would be connected to, or designed as an integral part of one of the two mates. The required contact pressure may be provided by the connector actuators or may be supplied as an integral part of the seal.

2.4.3.3 Final Concept Evaluation

As previously stated, the preliminary evaluation concluded in the choice of four concepts. These concepts and their representative drawings are tabulated below:

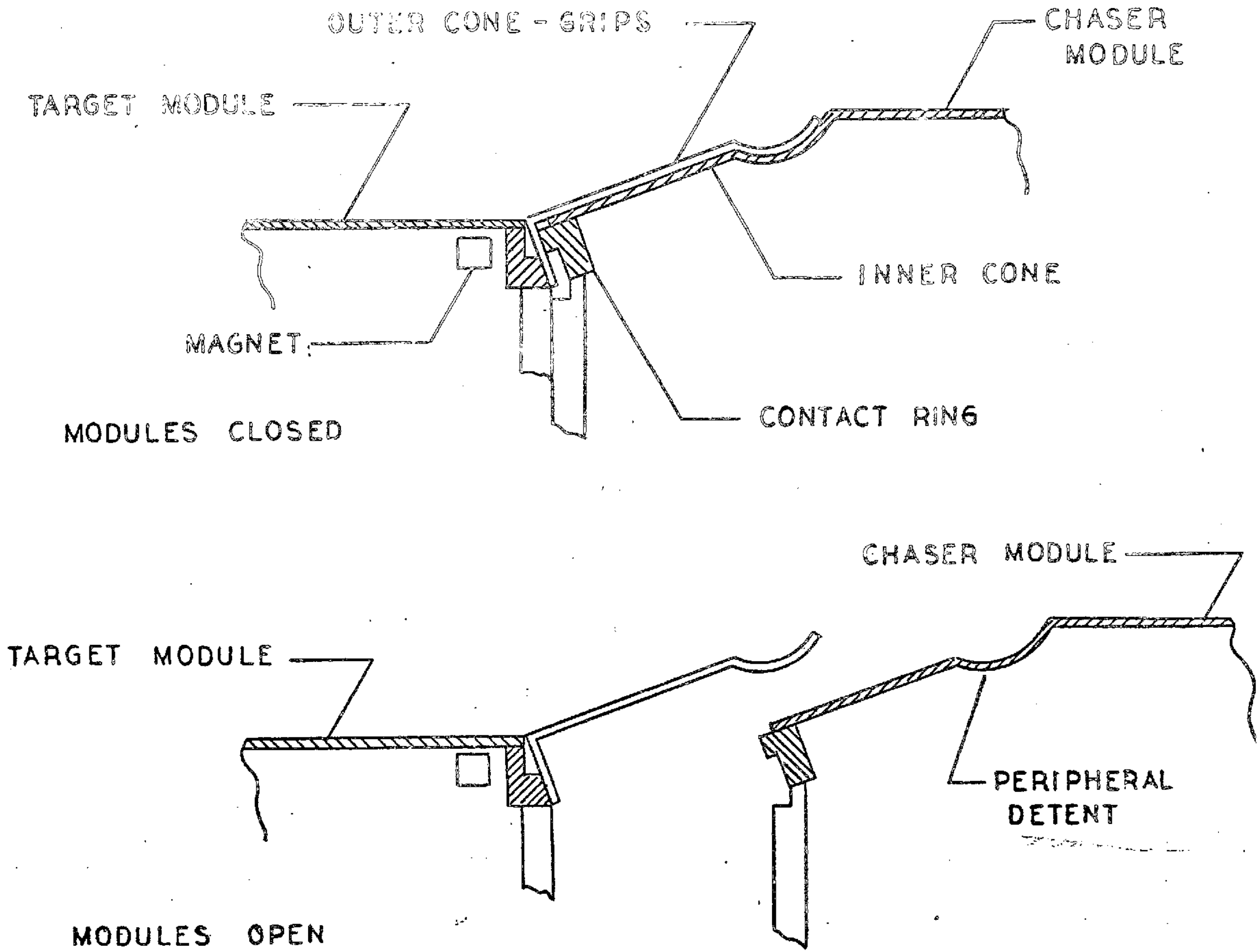
Concept	Figure No.
Conically Arranged Swivel Fastener Assembly	2 - 24

Concept (Cont'd)	Figure No.
Snap Wedge Assembly	2 - 26
Key Pin Assembly	2 - 27
Bell Crank Assembly	2 - 28

The final evaluation of these concepts is contained in figure 2-29, Docking and Assembly Concept Evaluation Chart. The initial chart contained 20 evaluation factors which were chosen and weighted with particular emphasis on the ability to demonstrate early mission capability and a high probability of mission success. As the evaluation progressed three items; safety to man, material availability, and time to effect operation were dropped from the chart since all the concepts being analysed were deemed equal in these respects. The choice of the rating criteria and their relative weights, and the actual concept evaluation represents the judgement of the three men most intimate with this area. In many cases the individual criteria score does not represent the choice of any of the three raters, but is the average value, taken when agreement could not be made. This method of evaluation has a great deal to offer. The continual reassesment of relative factor weights as the analysis progressed, and the deep probing into individual details stimulated by the desire to substantiate the individual initial scoring improved both the designs and the evaluations.

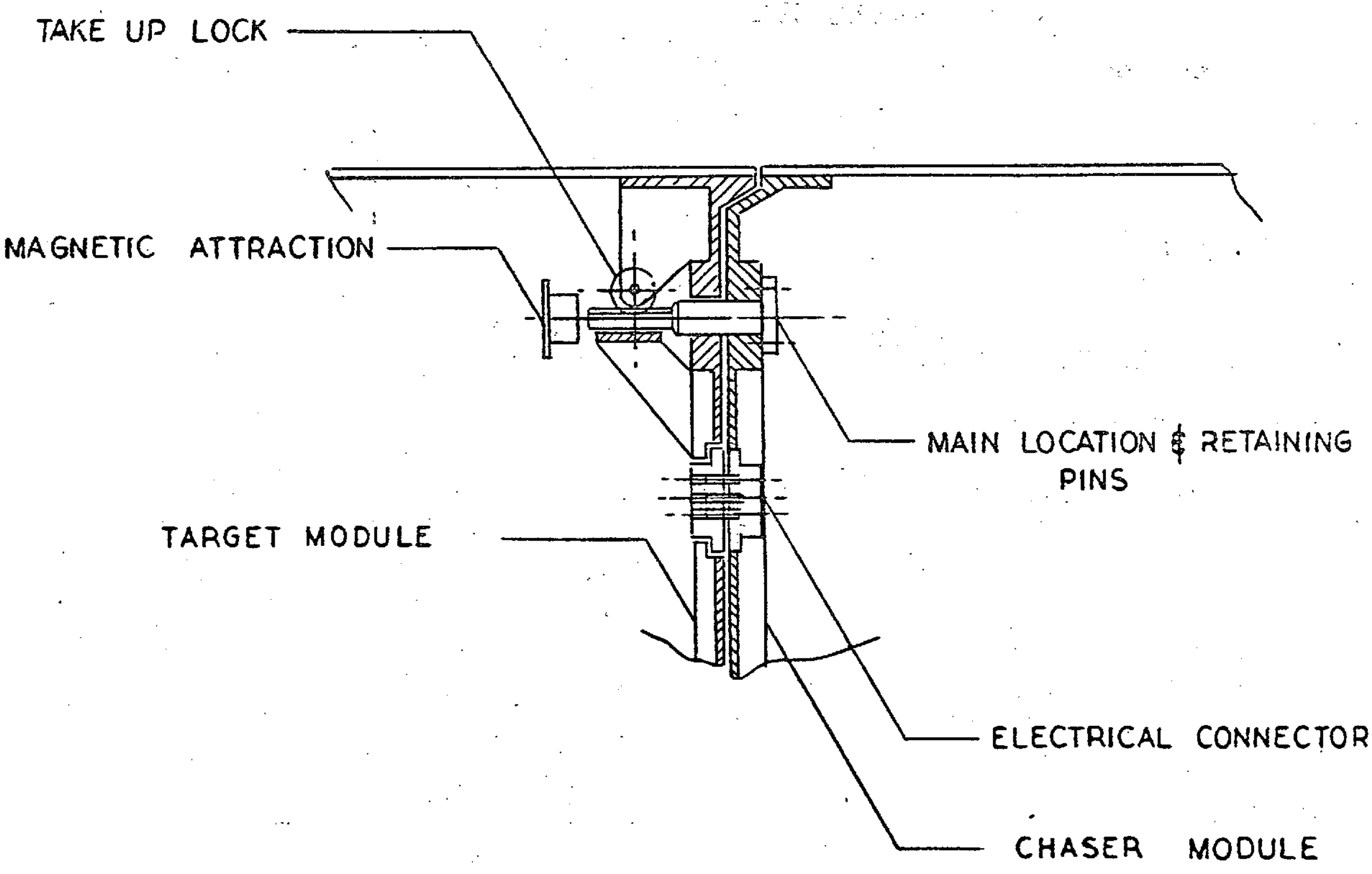
As can be seen in Figure 2-29, the criteria scores for the four concepts, prior to factoring by the relative weight, ranged from 10 to 7. In general the concept most desired was given a score of 10 and the remaining concepts were scored 9, 8, and 7 in relation to their desirability standing. This method was chosen rather than the 0% to 100% method since these four concepts represent the best of about 18 concepts initially investigated and it is therefore felt that they should all fall into the 70% or above category.

The results of the evaluation indicate that the Conically Arranged Swivel Fastener Assembly offers, by a narrow margin, the most promising method of performing the docking and assembly operation. To increase the confidence level in the choice of this technique, a development and testing program was set up. An analysis of this program with respect to feasibility, and accomplishment in sufficient time to perform the scheduled initial manned lunar mission was made. This analysis substantiated the evaluation since it was concluded that this concept offered a high probability of conversion into functioning hardware within the required time limitation. The salient points of this analysis are reflected in the program recommendations contained elsewhere in this section.

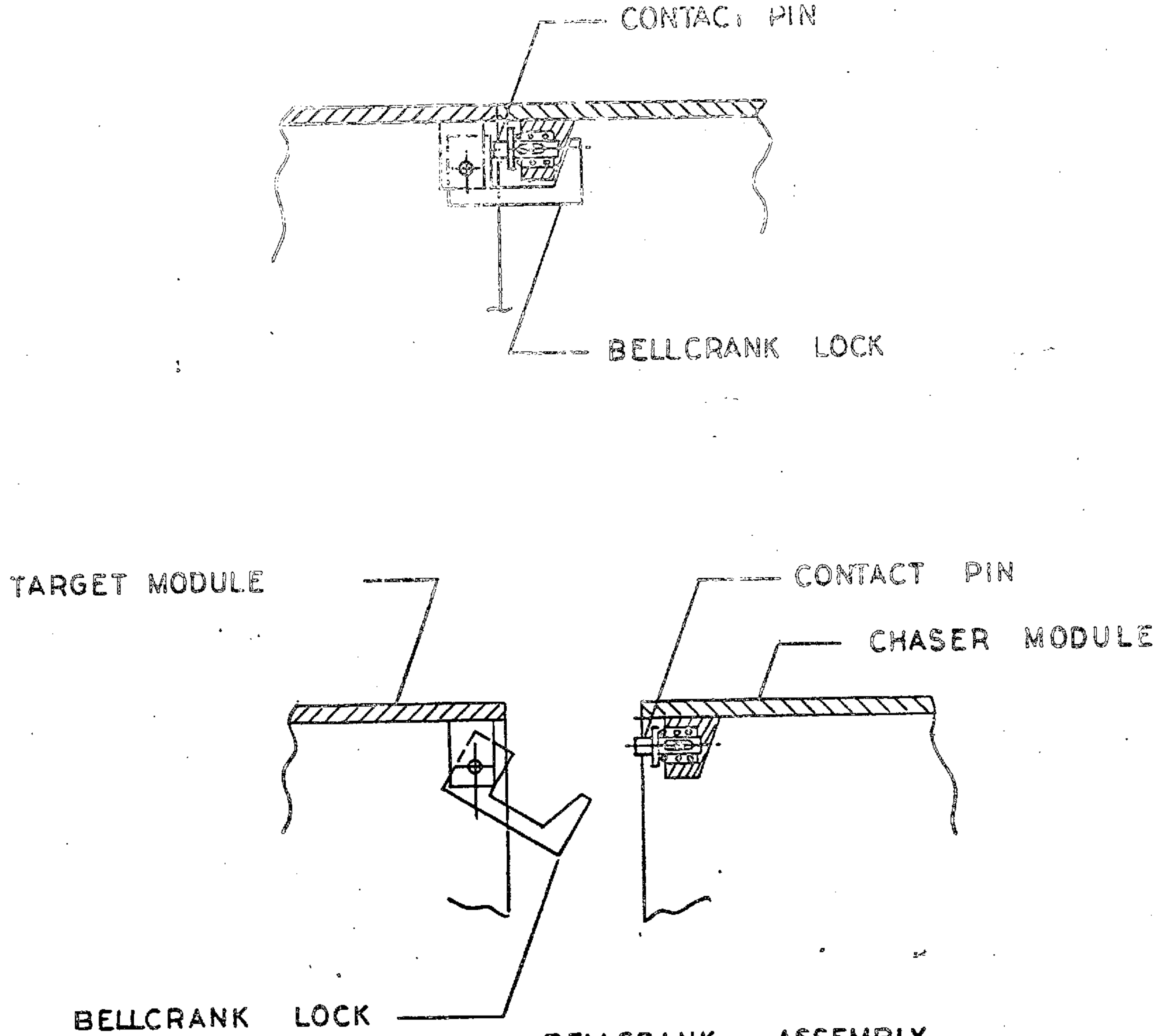


SNAP WEDGE ASSEMBLY

FIGURE 2-2[illegible]



KEY PIN ASSEMBLY
FIGURE 2 - 27



BELLCRANK ASSEMBLY
FIGURE 2-28

DOCKING AND ASSEMBLY CONCEPT EVALUATION

CRITERIA	WT. RATE	CONICAL ARR. SWIVEL FASTENER		SNAP WEDGE		KEY PIN		BELL CRANK	
1. Performance risk	10	10	100	9	90	8	80	7	70
2. Reliability	10	7	70	10	100	9	90	8	80
3. Development risk	10	10	100	7	70	8	80	9	90
4. Development time	9	8	72	7	63	10	90	9	81
5. Ability to detect malfunction	8	10	80	7	56	9	72	8	64
6. Ability to correct malfunctions	8	10	80	7	56	9	72	7	56
7. Flexibility	7	9	63	7	49	10	70	8	56
8. Necessity for external protective needs	7	9	63	10	70	8	56	8	56
9. Tolerance	7	10	70	8	56	7	49	9	63
10. Ability to connect and disconnect	6	10	60	7	42	9	54	7	42
11. Alignment accuracy	6	10	60	7	42	9	54	8	48
12. Req. storage space	6	8	48	7	42	9	54	10	60
13. Complexity	5	7	35	10	50	8	40	9	45
14. Power requirements	5	8	40	9	45	7	35	10	50
15. Weight of system	4	7	28	8	32	10	40	9	36
16. Development testing requirements	4	10	40	7	28	9	36	8	32
17. Development cost	4	9	36	7	28	8	32	10	40
TOTAL SCORE			1045		919		1004		969

FIGURE 2-29

2.4.4 Special System Component Tasks (Personnel and Propellant Transfer)

The special system component tasks of the Assembly operation include both the transfer of crew members between modules and the transfer of the propellant between modules. These tasks require special technologies of their own as well as technologies common to the other tasks involved. The AMF portion of this study is basically concerned with the mechanisms required to effect these transfers in orbit.

2.4.4.1 Crew Transfer

During this reporting period only a cursory study was made of the general requirements and constraints involving Crew Transfer. These were examined in light of the background of present knowledge regarding air locks, soft space suits, hard-shell capsules and Self-Maneuvering Units; and then against the types of concepts which have been proposed for retrieval, docking, and launching.

2.4.4.2 Propellant Transfer

(a) Task Definition

To determine the concepts and techniques needed for the orbital propellant transfer mode of OLO operations, and evaluate the effect of the various parameters on mission success. Then to establish the program necessary to develop and test the pacing components of these concepts, and integrate the hardware development program into the over-all milestone.

(b) General Criteria

1. Orbital Altitude 350-550 statute miles. Vacuum 10^{-9} mm Hg.

2. Length of time in orbit prior to transfer or launch - 30 days maximum.

3. The total weight of the S/V (Space Vehicle) at launching condition will be about 360,000 pounds. To ready the OLV for launch requires propellant load of approximately 180,000 pounds. The propellants to be used are liquid oxygen and liquid hydrogen. Non-lunar missions may use RP-1 in early tests.

4. The re-fueling of the OLV is to be accomplished in orbit from a tanker consisting of a modified S-IVB stage. The tanker can be boosted into orbit with fuel sufficient to re-fuel the OLV in a single transfer operation provided that storage and transfer losses are not excessive. The tanker module is to be the active member (chaser) in the rendezvous with the OLV.

5. The tanker module can be tracked-in under controlled conditions to contact or stand off any desired distance.

6. The target module (Space Vehicle) will normally be orientated in orbit so as to always present the same face to the earth. After retrieval and docking operations are performed, the combined tanker and Space Vehicle can be rotated at any desired velocity during propellant transfer. If it is necessary to rotate the tanker module during storage in orbit, i.e. prior to retrieval, it must be stopped and stabilized with respect to the earth prior to the retrieval operation.

7. The fluid couplings on both target and chaser can be located wherever convenient, either along the exterior circumference or at the periphery at either end.

8. The orbital plane for transfer operations make a 30-degree angle with the equatorial plane.

9. Fluid transfer is to be completed in less than 45 minutes (one-half revolution).

(c) Technology Required

In order to get a better understanding of the technological factors involved in propellant storage and propellant transfer (so that the concepts for performing the retrieval, docking, and assembly required for successful transfer would be realistically related to the problems involved) arrangements were made with Arthur D. Little, Inc., of Cambridge, Mass. to have them perform the necessary studies for the Handling of Cryogenic Propellants in Orbital Launch Operations on an unfunded basis. The assignments represent a continuation of the roles respectively played in both the Titan and Atlas programs by ADL and AMF.

The evaluations and calculations made to support the recommendations made by ADL were of necessity of a very preliminary nature. In spite of this qualification, however, it appears that the background of knowledge and engineering techniques relating to the handling of cryogenic propellants is sufficient to support a successful orbital re-fueling operation within the time span set for our manned exploration of the moon.

Recommendations and Conclusions

1. Storage

It is preferable to store the cryogenic propellants in space in a non-vented system. This type of storage minimizes the loss of usable propellant for a given weight of insulation and avoids the problem of venting in a zero-gravity situation. However, as dependent on the safe-working pressure for the tanks, on the effectiveness of the insulation applied to the storage vessels and on their stay time in space, the constant volume method of storage may have to give way to storage as a boiling liquid with venting.

In order to minimize the loss in mass of propellant during the storage period it is preferable to arrange the thermal balances on the fueled modules to preserve the oxygen without loss and have all the heat input to the propellants result in a loss of hydrogen. It must be recognized, however, that the propellant mass ratios will vary as a function of time due to loss of hydrogen and that planning of the re-fueling operation must take this loss into account.

The ideal tanker module from the standpoint of minimum loss of propellant would have a size just sufficient to hold the maximum amount of propellant that can be boosted into orbit. Such a tanker would have a minimum ratio of area-to-propellant weight and hence reduce the heat input from the environment per unit of propellant load. The initial size is the important point, not the propellant to volume ratio.

The loss of usable propellant (hydrogen) while stored for thirty days in a 450-mile parking orbit is estimated to be in the range of 0-240 lb/day for a fueled S-IV stage and 100 to 1000 lb/day for a partially-fueled S-II stage. The lower figures are representative of losses in a non-vented storage while the higher figures are representative of losses associated with a vented storage. The upper ends of the ranges given are considered more realistic targets for vessel-insulation combinations which can be developed in the short time span of two to three years available before design must begin.

While further consideration should be given to other methods for reducing the heat inleakage to the propellant storage vessels such as orientation of the vehicle system, use of cork or foam-type insulations, and the use of a limited number of reflecting shields; it is unlikely that any scheme other than the use of super insulations can meet the requirements of OLO. Accordingly, we recommend that developments necessary to make super-insulated vessels for the space storage of cryogenic propellants a practical reality be given top priority.

2. Transfer

The separation of the liquid and gas phases and their positive location we regard as a fundamental requirement for propellant transfer. In our opinion the most promising method for meeting this requirement is to affect a fixed end-to-end coupling of the active and passive modules and to rotate this system about its common longitudinal axis.

The use of acceleration fields for developing pumping pressures is not practical. In the linearly accelerated system reasonable pumping pressures of the order of 10 psi are developed only with excessive "g" fields. Acceleration fields developed by rotation of the system produce pumping pressure in the desired direction for a portion of the transfer only, or not at all. The basic problem is that the head disappears when the transfer is half completed. The indication is that pumps would be needed in conjunction with such rotation.

Gas pressurized transfer is a current practice used both in ground and missile systems for the transfer of cryogenic liquids. In the orbital transfer operation this concept is both feasible and practical if the liquid and gas phases of each propellant are separated and stabilized with regard to location in the propellant tank. It appears that less than 1 g would be required to accomplish this. Perhaps it might be accomplished around .1 to .2 g.

If transfer of hydrogen and oxygen is to be accomplished by the use of gas stored in bottles at high pressure, then helium is the proper choice of pressurant and it should be cold-stored in the propellants.

Pressurization utilizing "wet accumulator action", i.e., self-pressurization of the propellants by vaporization within the tanks is practical and can be used for liquid transfer. Minimum propellant losses due to flashing liquid occur when the tank pressure of the active module exceeds the passive module pressure. Due to its simplicity this system is potentially more reliable than other methods of transfer.

The concept of positive propellant isolation and expulsion through the use of bladders, pistons or bellows appears feasible for orbital transfer operations. Bladders appear to be the most practical type of positive displacement device. Various bladder materials have shown promising results when used for the expulsion of cryogens from small static tanks; however, serious doubts still exist as to the reliability of bladders when subjected to slosh loads in large tanks.

The use of pumps for the orbital transfer operation is practical and may show considerable weight advantage over a gas pressurization system if these devices have a low (ten feet or less) NPSH requirement under the specific operating conditions. Otherwise, the gas pressurization system needed to supply the necessary NPSH will degrade the reliability of the transfer system compared to that of the gas pressurization system alone and it will compromise the inherent weight advantage of the pump.

3. Measurement

A continuous record of the liquid propellant mass in each tank of the active and passive modules will be required in order to comply with mission prediction and logistic needs. The mass can probably be determined indirectly from independent measurements of the propellant properties, from the known characteristics of the tank configuration, and from knowledge of the thermodynamic processes at all points of the boost and orbit regimes.

A concept for establishing mass quantities involves the flow measurement of all the fluids entering and leaving the propellant tanks and the continuous measurement of the independent properties of the propellants such as volume and pressure. Automatic computations would be used to derive and compare mass quantities resulting from at least two independent methods. Heat and mass balances would be performed on each propellant in its system. When acceptable correspondence between independent mass measurements are not obtained the differences in the measuring technique are exploited to troubleshoot and isolate propellant and measurement system failures.

Wherever possible, the instruments and techniques used in current practice should be applied to the problem of mass measurement. Whenever the propellant tank systems are accelerated to the degree required to positively stabilize and locate the liquids within them, measurement of "liquid level" by the techniques in current practice can be used as a basis for a mass determination. The acceleration response of a system to a known thrust is another obvious means for inferring propellant mass that bears some further attention.

In a zero "g" environment the propellant and its tank can be treated as an isolated thermodynamic system. By assuring isothermal conditions the total fluid mass and the quality of the mixture can be determined from knowledge of the thermodynamic properties of the system and its pressure response to a known heat input. Although the application of basic thermodynamics as a method for determining propellant mass is well grounded in theory, the accuracy of this measurement technique has yet to be assessed. This technique may develop into a useful tool for identifying failures and malfunctions in the propellant system.

A tesserae capacitance structure can, in concept, be used to determine liquid mass from a single measured quantity.

4. Integrated Storage and Transfer Concept

The following concept is suggested as a basis for meeting the requirements for in-orbit fueling associated with Orbital Launch Operations.

Both modules should incorporate a thermal protection system based on the multi-shield reflecting type of insulation. In preparation for the transfer of propellants from the tanker to the OLV, the tanker is joined with the OLV in a fixed end-to-end configuration as shown in Figure 2-1. The propellant transfer lines between the active and passive modules are coupled in consonance with the major joining operation. Immediately prior to and during transfer an artificial force field is generated by acceleration of the coupled system either linearly in the direction of the common axis or by rotation about this common axis.

The motion of the system would be induced by reaction jets. The energy required to achieve and maintain a centripetal acceleration field is less than that needed to maintain a linear acceleration.

With the propellants positively located and stabilized in the system, conventional means are used to transfer the propellants from the tanker to the OLV and to meter the process. Pumps, helium gas pressurization, and a wet accumulator action are all practical methods for producing the transfer in this situation and, moreover, they have the advantage of having been reduced to practice in ground handling and missile systems. The choice of the best transfer method must await further evaluations.

(d) Propellant Transfer Coupling Mechanism Concepts

Concepts for fuel transfer fall into the two general categories:

1. Flexible hose concepts.

2. Rigid connection concepts.

In the flexible hose concepts the modules are physically connected only by the propellant hoses. Flexible concepts may be further broken down into two sub-groups:

a. Using a "flying hose". See Figure 2-30.

b. Using booms or extendable arms. See Figure 2-31.

Rigid connection concepts are characterized by a firm physical connection between the modules. More specifically, the chaser and target have gone through the operations of retrieval, docking, and assembly. Rigid concepts may also be sub-divided into two groups:

a. Re-usable tanker concepts (they can be returned to earth). See Figure 2-32.

b. Concepts where the chaser becomes an integral part of the target, and serves as part or all of the fuel tank. Since true propellant transfer does not take place here, this concept is basically an assembly operation.

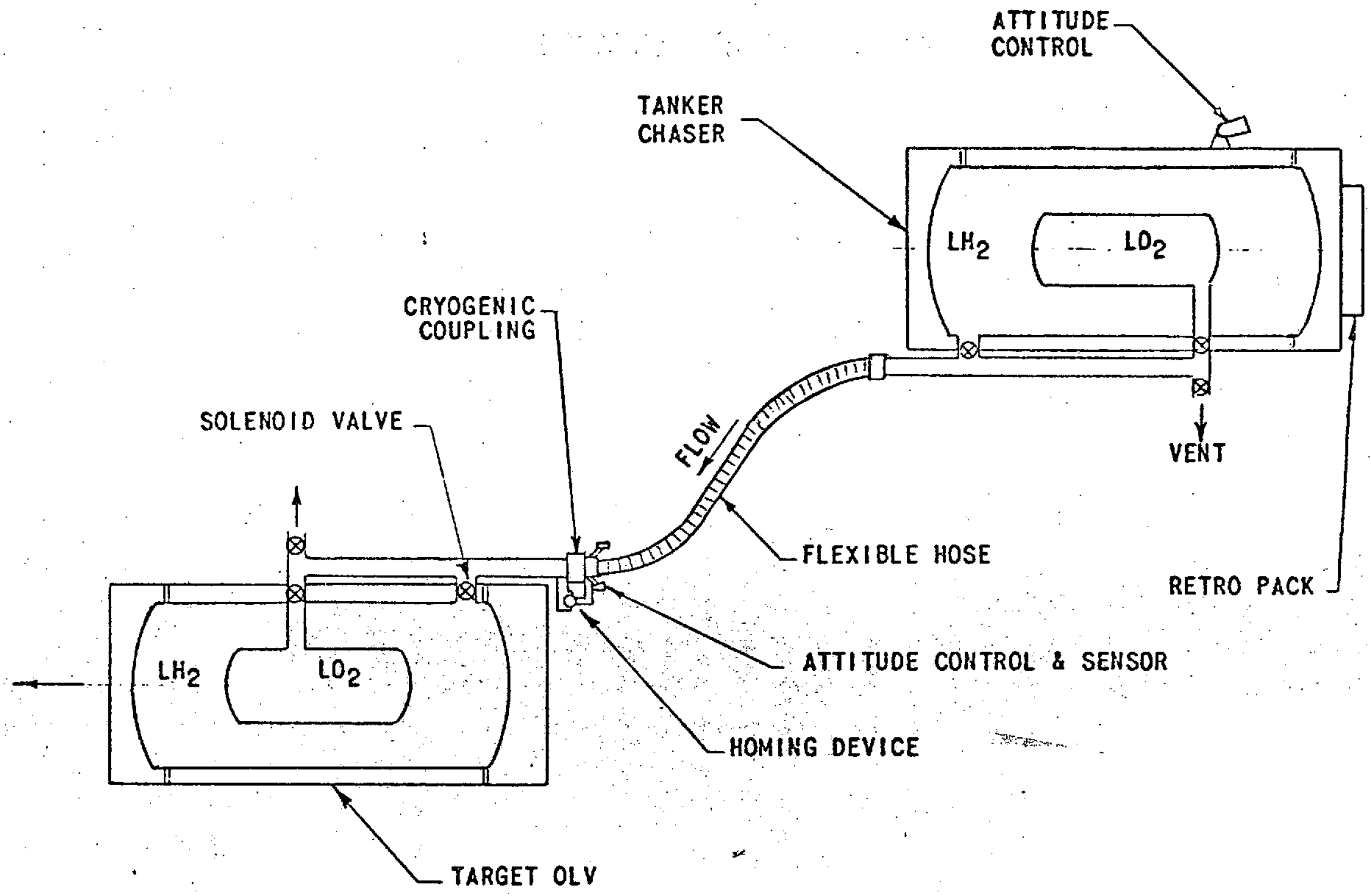
There were other concepts proposed and examined which were eliminated during the preliminary evaluation. Most of these eliminated concepts fall into the "blue sky" category. The following are some of such concepts eliminated:

a. Harpoon Concept

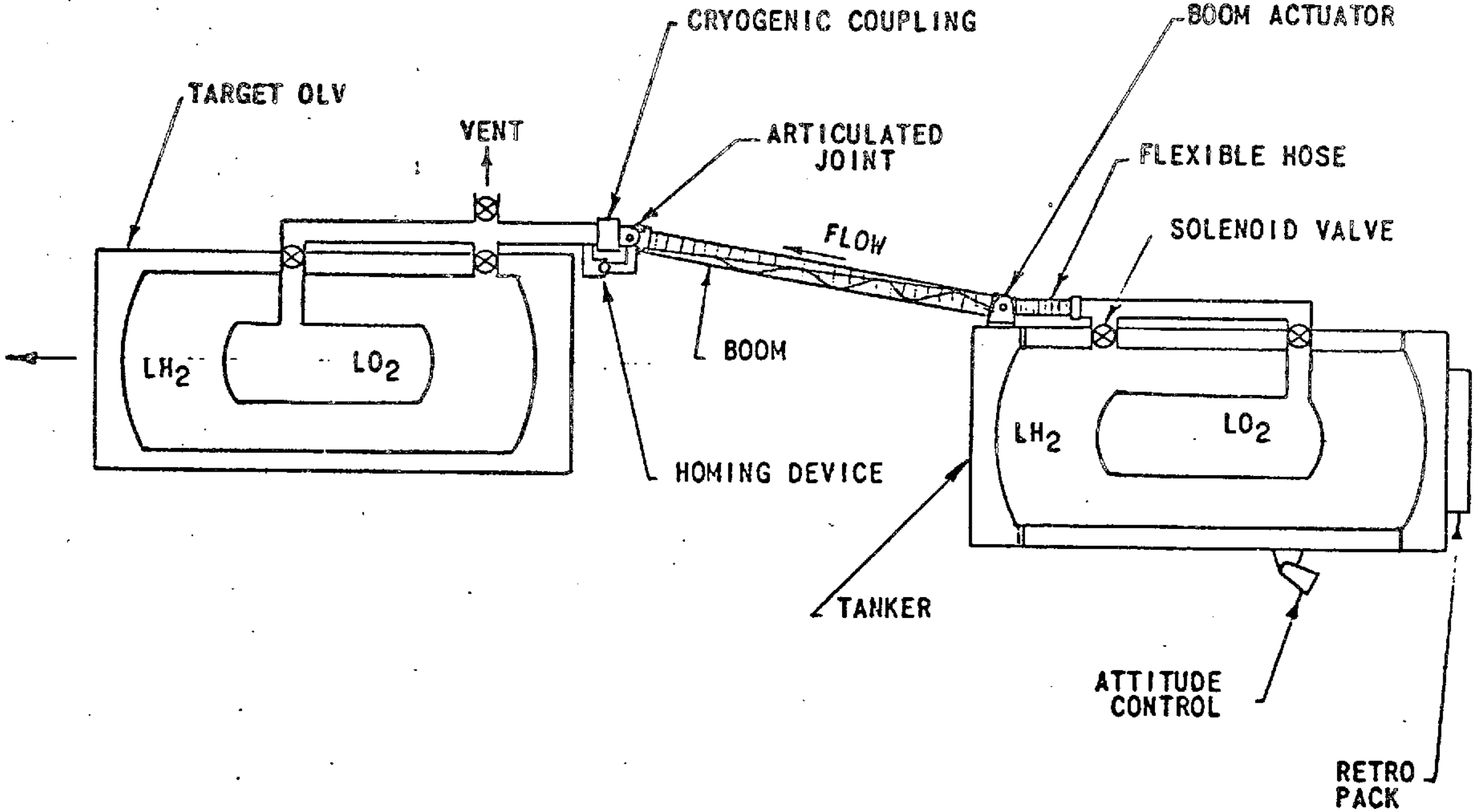
The fuel line is shot across to the target from the chaser by a mechanism similar to a harpoon gun. The gun is aimed by a sensing mechanism. The cryogenic coupling at the end of the harpoon, penetrates the empty fuel tank and seals itself. After the seal is inspected automatically, a signal starts the propellant transfer.

b. Inflatable Hose Concept.

The hose is coiled in such a manner that upon admittance of gaseous pressure, the hose extends linearly. When extended the hose contacts the module to be fueled, causing the cryogenic couplings to mate.

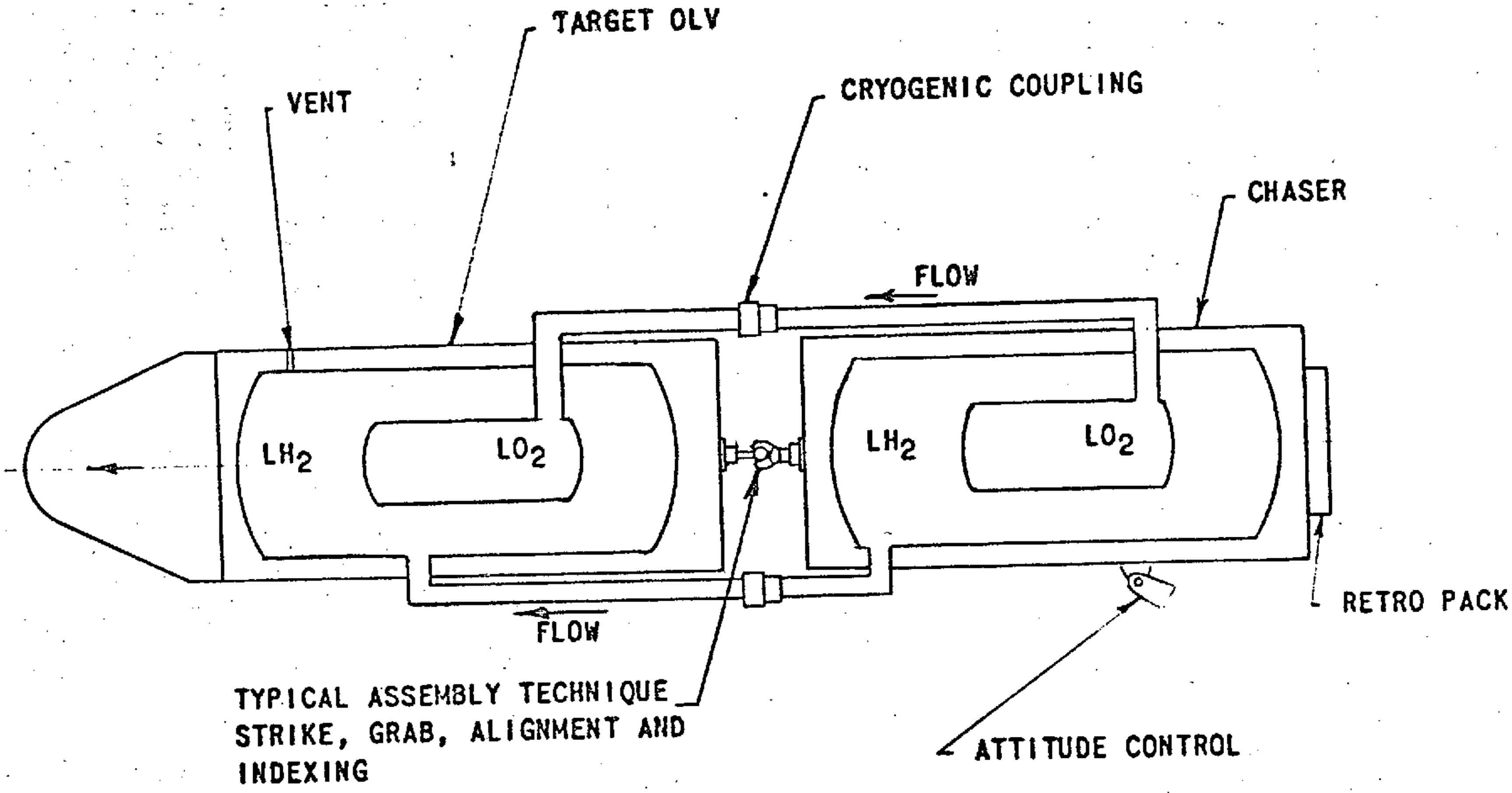


FLYING HOSE CONCEPT
FIGURE 2-30



PROPELLANT BOOM CONCEPT

FIGURE 2-31



RE-USABLE TANKER MODULE CONCEPT

FIGURE 2-32

Cryogenic couplings shown earlier in Figure 2-23 are common to all propellant concepts.

A drawing of a typical coupling for LH_2 obtained from On-Mark Corp. (Reference their Drawing No. 8173) shows its temperature range as $-423^{\circ}F$ to $+250^{\circ}F$, at 200 psi operating pressure. Basically, it consists of two halves, each with a check valve. When the two halves are apart, the check valves are closed permitting no leakage. When engaged, the check valves open, permitting fluid passage while at the same instant sealing the coupling against external leakage. Fingers lock the two halves together. The couplings can be separated pneumatically, mechanically, or manually. Sizes range from 1-1/2 inches to 10 inches, and both halves together, weigh from 7.2 to 109.7 lbs. The materials used in their construction are basically aluminum and stainless steel. A typical 6 inch coupling can deliver 5000 GPM.

Some of the features desired in a cryogenic coupling are:

1. No leakage in each half before joining, during fuel transfer, and after separation.

2. Ability to mate under conditions of imperfect alignment and orientation.

3. Ability to detect, and signal an imperfect connection.

4. Ability to correct an imperfect connection.

5. Require little force to effect joining and separation.

6. Light weight.

(e) Evaluation of Propellant Transfer Coupling Concepts

1. All flexible hose or boom concepts require some means of sensing to bring the two coupling halves together. Heat, light, radar, and magnetism are possible means that can be used. The flexible hose could be a bellows type, stainless steel construction with teflon braided covering similar to the type currently used on earth for cryogenic commodities.

In the concept using booms or extendable arms, a sensing device is employed at the end of a boom or linkage. The boom supports the flexible hose and connector. In operation the boom is guided into position when the target and the chaser are in close proximity, by a sensing device on the target. After the two coupling halves strike initially, a rough alignment takes place. Then a mechanism engages both halves and makes the final alignment and mating. The main advantage this concept has to offer is its ability to fuel various modules with propellant ports located in different positions. It has many disadvantages, the most glaring being complication and weight. The boom, and actuators for the boom, would add considerable weight to the system.

In trying to overcome the weight problem which this concept poses, the "flying hose" concept emerged. In the "flying hose" concept, no booms are used. The hose is wrapped either externally around the periphery of the chaser, or stored internally on a reel. At the end of the hose is a cryogenic coupling, which has three attitude control rockets. A sensing device is also incorporated at this end. When the chaser and the target are in close proximity, the hose is payed out from the chaser, and guided into position by the sensor on the target. From this point on, the action is similar to the previous concept. The principal advantage this concept has to offer is lighter weight, by eliminating the boom or linkage.

Both flexible concepts breakdown under closer analysis, and must be removed from consideration. Strike, grab, alignment, indexing, etc., must take place as in the case of the rigid schemes. The only advantage flexible hose concepts offer is versatility or ability to fuel many types of modules, whose propellant ports may be located differently. The fluid dynamics involved with flexible lines is another negative factor in such concepts.

2. For all rigid connection concepts, it is assumed the modules have already been brought together. Strike, grab, alignment, indexing etc., have already taken place. The only thing to be considered is coupling of the propellant lines. In the general concept (see Figure 2-32) the chaser is reusable and can be recovered after propellant transfer. In operation after the two modules are together, or during the final phase of coming together, the cryogenic couplings are mated. This mating of couplings will take place after an initial rough alignment. A mechanism will effect the final alignment and mating. A signal will determine whether or not the connection is faulty. If a faulty connection has taken place, the coupling mechanism will be reversed and a new coupling effected. After the propellant transfer is completed, the chaser is separated from the target, and returned to earth by retro-rocket and parachute sequence.

The re-usable chaser concept requires only partial assembly, and the union of cryogenic couplings. It is also simplest, from the viewpoint of operations necessary to effect propellant transfer. The purpose of this report, is not to advocate, or suggest that the propellant transfer be adopted. However, if propellant transfer is to be carried out, it appears that the best approach would be the reusable chaser concept.

3. A point that should not be overlooked in evaluating the relative merits of propellant transfer versus those of orbital assembly is the increased reliability of the space-vehicle attained. Since the vehicle utilizes a completely earth assembled and checked out system, its reliability should be higher than that of a vehicle with space assembled connections.

2.5 USE OF MAN AND OLF

2.5.1 Basic Criteria

A major problem area pertinent to this study is the determination of the interrelated effects posed by the use of man, the use of an OLF, and the allocation of specific tasks between man and automatic equipment. In order to guide the direction of work performed during this study the following man-machine relationships have been adopted:

(a) Automatic - This operation requires no operational man functions except for monitoring. System control is performed from earth stations.

(b) Automatic with Man - This operation is identical to an automatic operation except that since man is aboard he monitors certain automatic functions and is given an overriding control option. Man is a redundant to these functions.

(c) Semi-Automatic - In this operation man in integrated into the loop as a series function. He replaces automatic machine operations where his superior capabilities enhance the mission requirements.

(d) Manual Operation - An integrated man-machine system where man is the dominant controlling element. Functional commitment is solely dependent on man's discretion and aids. Many operations are performed manually.

2.5.1.1 Criteria

In addition to the man-machine relationships described above, the following criteria served as a basis for performing the detailed analysis.

(a) Man and automatic equipment are viewed as having capabilities that are complementary--not competitive.

(b) The main objectives of the program are; high probability of mission success (reliability), crew safety, and early development and demonstration of a proven capability.

(c) Superior abilities of man and automatic equipment are:

Man	Machine
Decision making	Application of large forces
Inductive reasoning and judgment	Deductive computations
On-the-spot programming (i.e., cope with unexpected events).	Error free repetitive task performance
Improvise and use flexible procedures	Rapid response to control signals

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Man	Machine
Maintenance and repair	Performance of many highly complex tasks in short time period
No constant physical tie-in required	Precision
Affords system "graceful degradation" in performance in lieu of catastrophic failure	

(d) Man's ability to be useful in a spacesuit must be proven.

(e) Man must be trained to accomplish tasks under circumstances which cannot be accurately simulated on earth.

(f) Man's weight, and the weight of his life-support equipment must be considered in a trade-off with automatic equipment.

(g) When man is present and has the time and ability to perform, he should be used.

(h) Man's learning curve is much longer and has a much larger variation than that for automatic equipment.

(i) Man requires higher booster reliability and confidence level--therefore longer rating time.

2.5.2 Approach

The basic approach adopted for the performance of this study is described elsewhere in this report. It includes a determination and documentation of basic criteria, a mission analysis, a function analysis, and a task analysis.

2.5.3 Analysis and Conclusions

2.5.2.1 OLF Analysis Results

An analysis was performed to determine the minimum OLF requirements to satisfy the four man-machine relationships adopted; automatic, automatic with man, semi-automatic, and manned. Since the basis for selection was restricted to early accomplishment of the initial manned lunar mission, and in light of the mechanism concepts evaluated, it was concluded that all assembly operations could be performed without the use of an OLF. However, the operation may benefit from the use of a minimum OLF. This minimum OLF is comprised of an Apollo (or similar type capsule) with a mission module outfitted on earth and containing assembly, check-out and launch support equipment.

2.5.4 Man-Machine Analysis Results

Automatic, automatic with man, semi-automatic and manual modes of operation were analyzed and resulted in the conclusion that the "automatic with man" mode of operation best enhanced the goal of early mission success probability.

A large variety of possible manual concepts were investigated. These concepts all suffered from one or a combination of the following limitations:

1. Requirements imposed by life support and crew safety delays the time at which tests of new systems can be made. Since the initial lunar mission will be demonstrated through a progressive build-up of capabilities, the integrated time delays will become appreciable. An example of the basic delay factor is well illustrated in our Mercury Program which uses an automatic shot, and then a primate shot, prior to allowing a manned shot for each progressive increased capability.

2. In a space-vehicle in general, direct observation by man is difficult and often dangerous. Such devices as television, periscopes, and optical rods must be used together with filters, etc., all of which detract from his sensing ability. When one considers the closeness of the modules in assembly operations, the matter becomes even worse. Bulky modules and the simplified concepts which require rear-of-target to front-of-chaser type connections make both instrument and direct observation over the long distances extremely complicated. Even side locations for man, which can be attained by use of the minimal OLF, would suffer from blind spots.

3. As previously mentioned in the basic criteria, man's learning process is generally much longer than that of a machine. The time required for man to achieve a sufficient proficiency to demonstrate a reasonable reliability detracts from his direct use. The reliability curves developed later in this section substantiate this opinion.

4. Although man eliminates some equipment, usually servo systems and amplifiers, he normally still requires external sensor inputs to a display.

5. Placing man in an outside position as an operator, controller, or observer in a space suit or light weight capsule is extremely dangerous when one considers the large packages involved and their corresponding inertial forces.

The above factors, together with other factors not listed, and the list of basic criteria previously developed, lead to the conclusion that the manual and semi-automatic modes probably do not afford advantages when compared with the "automatic with man" mode, for orbital assembly operations. The automatic mode is considerably enhanced by the use of manned assistance as can be seen by the reliability curves, Figures 2-33, 2-34, and 2-35.

2.6 Weight Analysis Summary

The preliminary weight estimates for the chosen retrieval concept, and docking and assembly concept are:

Rigid Arm - Side Location	740#
Conically Arranged Swivel Fastener Assembly-	2,100#

These values are those required at each interface and are partially provided on each of the two mating modules. The weight values were presented to the program coordinator who integrated them with other sub-system contributions in order to prove the feasibility of the total system with respect to weight effects.

It is worth mentioning that appreciable weight savings, in the form of reduced structure, are gained by using side mounting rather than center mounting for the retrieval and assembly hardware.

The 2,100 pound weight for the Conically Arranged Swivel Fastener Concept includes; structural members and stiffeners, mechanical fasteners and hardware, interface connectors, actuators, solonoids, seals, and shrouds to protect the assemblies while in space storage. The estimate is based on the use of eight assemblies placed on the peripherey at the intersection of 45° radial lines.

The 740 pound weight for the Rigid Boom - Side Location includes the necessary structures and stiffeners, boom weights, mechanisms, and drives, sensors, controls, and hardware. The weights are based on a design which allows a maximum impact at initial strike, with booms fully extended, of about 3000 lb-sec. The maximum boom axial force is prevented from building above the safe level by the provision of a slip clutch & energy absorption mechanism. Preliminary calculations indicate that about a 2 second slip time will be sufficient to provide adequate energy absorption and retain the boom axial load within a safe value.

2.7 Schedule and Cost Summary

Schedules are presented for the design, qualification testing, and fabrication of 30 production units for:

(a) Rigid Boom (Side Location) Retrieval Mechanisms (Figure 2-36)

(b) Conically Arranged Swivel Fasteners - Docking and Assembly Mechanism (Figure 2-37)

(c) Fluid Coupling - for Propellant Transfer Assembly (Figure 2-38)

The cost data for the above program is presented in the summary below. These costs do not include launch vehicle costs.

COST SUMMARY

Item No.	Description	Cost*
1.	Retrieval - Rigid Boom - Side Location	$3,804,500**
2.	Assembly - Conically Arranged Swivel Fasteners	$3,835,500**
3.	Propellant Transfer - Assembly & Special Fluid Coupling	$3,104,500***

* All Costs Include 30 Production Units Shipped to Space Vehicle Manufacturer.

** The Costs for Three Atlas - Gemini Orbital Test Shots are not Included.

*** The Costs of Propellant Transfer Equipment is not Included.

2.8 Mission Success Probability

A detailed analysis was made to determine the components of each of the sub-systems which had a high rating in the evaluation analysis. Typi[illegible] detail of the analysis is the data contained in Table 2-2 for just the mechanism of the Rigid Boom - Side Location concept. (Common items to the various sub-systems such as sensors, amplifiers, controls, etc. were similarly treated by a separate analysis and were incorporated into the reliability and mission success probability computations at the appropriate point when various system configurations were postulated.) A summary of the better rated concepts is shown in Table 2-3.

System concepts were then constructed for the following types of operations:

a. Completely Automatic in orbit. If man serves any purpose, it is to press a button on earth to signal for continuation of a process.

b. Man Assisted Automatic - here the process is automatic, but man is aboard to monitor operations, take over controls in case of failure, make substitution type repairs, etc. Man serves mainly as a redundant loop to the sensors and amplifiers.

c. Semi-Automatic - here the process is divided to form the most complementary trade off between man and machine. Man furnishes certain of the sensors, others are accomplished by transducers which feed into displays for the man to observe. Man operates the simpler controls. Machines do the necessary computations either mechanically or electronically.

d. Manual - here the process is almost completely by man. Where required, the necessary computations are done by machine. Most procedures are controlled by man by levers or push-buttons, but the operations are carried out by some type of electro-mechanical device.

To obtain the growth rate figures, certain basic assumptions were made. Among the most important of these were: the equipment failure rate decreases by 25% each year; man's initial reliability is 50% and has an exponential increase to an ultimate reliability of 90% by 1967; the exponential failure rate law applies to both man and equipment; the mean time to failure for the sensors is 1200 hours (this is probably an extremely conservative estimate).

TABLE 2-2

FAILURE PROBABILITY ANALYSIS - MODULE RETRIEVING METHODS

CRITICAL COMPONENT	QUAN.	GFR[1] Fail. /10^6 Hrs.	KOP[2]	MISSION OPERATING TIME[3] Hrs.	MISSION FAIL. RATE Fail./ 10^6 Missions
A. <u>SIDE MOUNTED RIGID ARM MECHANISM</u>					
Lead Screw	2	.12	800	.167	32.1
Turntable	2	1.00	"	"	268.0
Worm Gear	2	.12	"	"	32.1
Bearings	12	.50	"	"	800.0
Electric Motors	4	.30	"	"	160.0
Arm Position Switches	12	.50	"	"	800.0
Grabbing Arms	3	.10	"	"	40.0
Indexing & Alignment Arms	3	.10	"	"	40.0
Pin Shafts	6	.03	"	"	24.1
Explosive Thrusters	2	.21	"	"	56.2
Limit Switch Trigger	1	.25	"	"	33.4
Control Relays	12	.30	"	"	482.0
Field Resistors	8	.04	"	"	42.8
					Total = 2,810.7 Fail. /10^6 Missions
					Reliability = .9972

Notes:

1 Generic Failure Rate Data was compiled from Martin Denver Document #MI-60-54 (Rev. I) and Martin Baltimore Reliability Handbook dated July 1959.

2 Operating Mode Factor (KOP = 800) was taken from Martin Denver Document #MI-60-54 (Rev. I) dated July 1961 and is the value which corresponds to nose cone compartment (in flight).

3 Design Parameters specified a mission operating time of 10 min. or .167 hrs.

TABLE 2-3

FAILURE PROBABILITY ANALYSIS SUMMARY

SYSTEM	TOTAL FAILURE RATE (FAILURES/10^6 MISSIONS)	MEAN TIME BETWEEN FAILURE (MISSIONS/FAILURE)	% RELIABILITY
I. Module Retrieving Methods			
*A. Side Mounted Rigid Arm Mech.	2,810.7	358	99.72
B. Center Mounted Rigid Arm Mech.	2,591.9	386	99.74
C. Flexible Line Passive Dart Mech.	377.7	2,650	99.96
II. Module Fastening & Separation Methods			
*A. Conically Arranged Swivel Fasteners			
Fastening	2,223.8	450	99.777
Separating	224.8	4,450	99.978
B. Snap Wedge Assembly			
Fastening	211.0	4,730	99.979
Separating	224.8	4,450	99.978
C. Bell Crank System			
Fastening	1,647.0	607	99.835
Separating	224.8	4,450	99.978
D. Key Pin Assembly			
Fastening	1,577.0	634	99.842
Separating	224.8	4,450	99.978
III. Propellant Transfer Methods**			
A. Re-usable Tanker Module Concept***			
Fastening	3,146.6	318	99.685
Separating	224.8	4,450	99.978
B. Flexible Hose Boom Concept			
Fastening & Separating	3,495.3	286	99.65

*Concepts used in construction of the best integrated system for highest mission success probability.

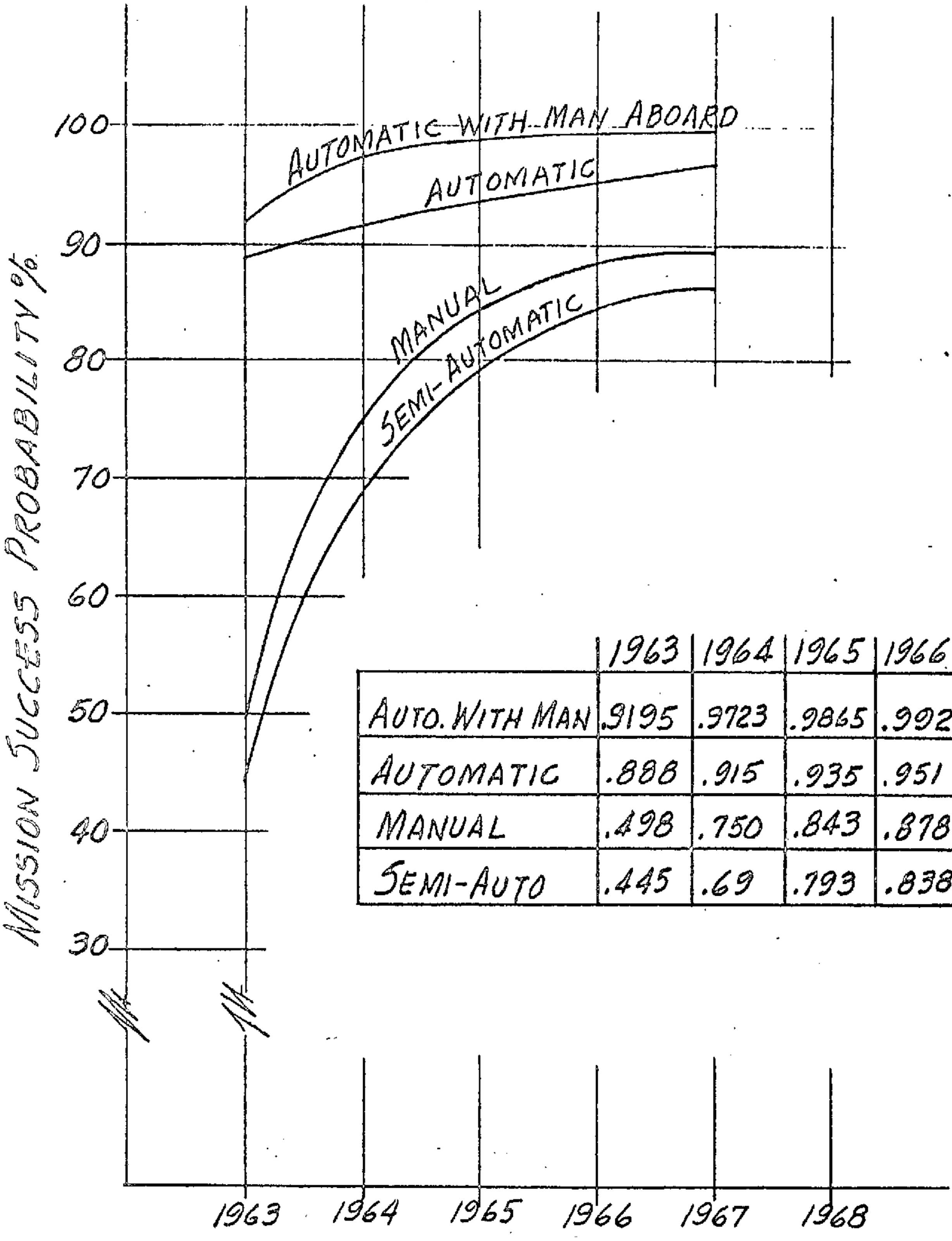
**Does not include means for propellant transfer but only for connection mechanisms, i.e., transfer itself is taken as unity.

***Method used in reliability growth analysis

The results of the Reliability Growth Analysis for the four types of situations are shown in Figures 2-33 and 2-34 for a two module operation. As will be noticed, the curves for Retrieval, Docking, and Assembly are very similiar to those for Retrieval, Docking, Assembly, and Propellant Transfer. This is occassioned by two principal factors: first, the Propellant Transfer values were calculated using unity for the process of obtaining the movement of and measurement of the fluids involved; and second, the complexity and failure rates for those assembly tasks not required in Propellant Transfer closely approximate the new components required for the transfer connections. The inference that must be drawn from this data is that Propellant Transfer will always have a lower probability for mission success than straight Assembly, since it is improbable that the equipment needed for the selected transfer technique will have an inherent reliability approaching .997. Other factors involved in mission accomplishment must be weighed against this transfer penalty to determine the best trade-off in each situation.

Where the operations of retrieval, docking, and assembly are repeated a number of times (when the Space Vehicle is composed of more than two modules), there is a rather rapid decrease in mission success probability for the automatic and semi-automatic modes of operation. The results of an analysis of the effect of requiring two, three, and four modules in 1967 is shown in Figures 2-35.

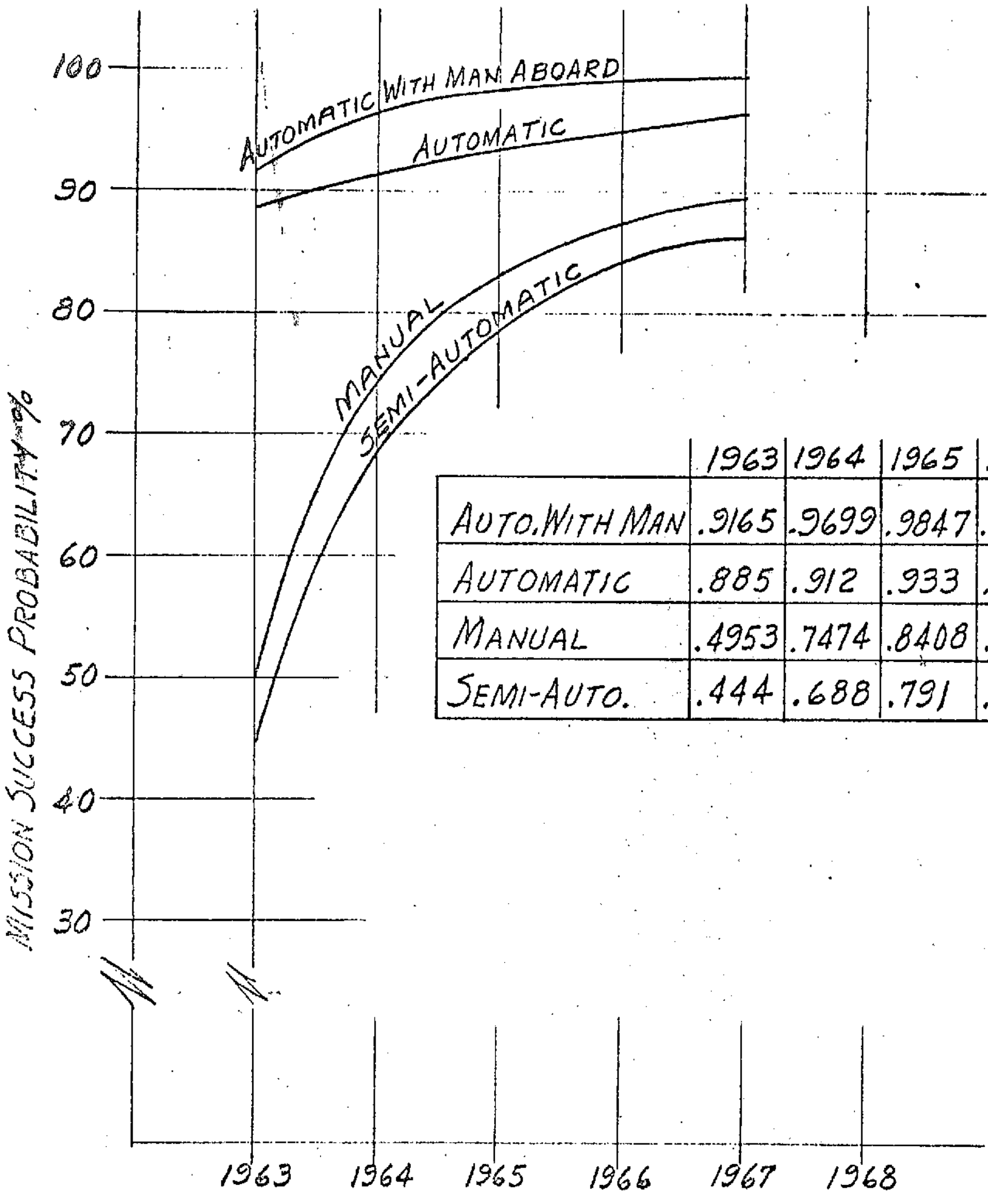
It is apparent from these studies that the Man Assited Automatic system of operation is the preferred system for the time period considered.



	1963	1964	1965	1966	1967
AUTO. WITH MAN	.9195	.9723	.9865	.992	.994
AUTOMATIC	.888	.915	.935	.951	.963
MANUAL	.498	.750	.843	.878	.894
SEMI-AUTO	.445	.69	.793	.838	.864

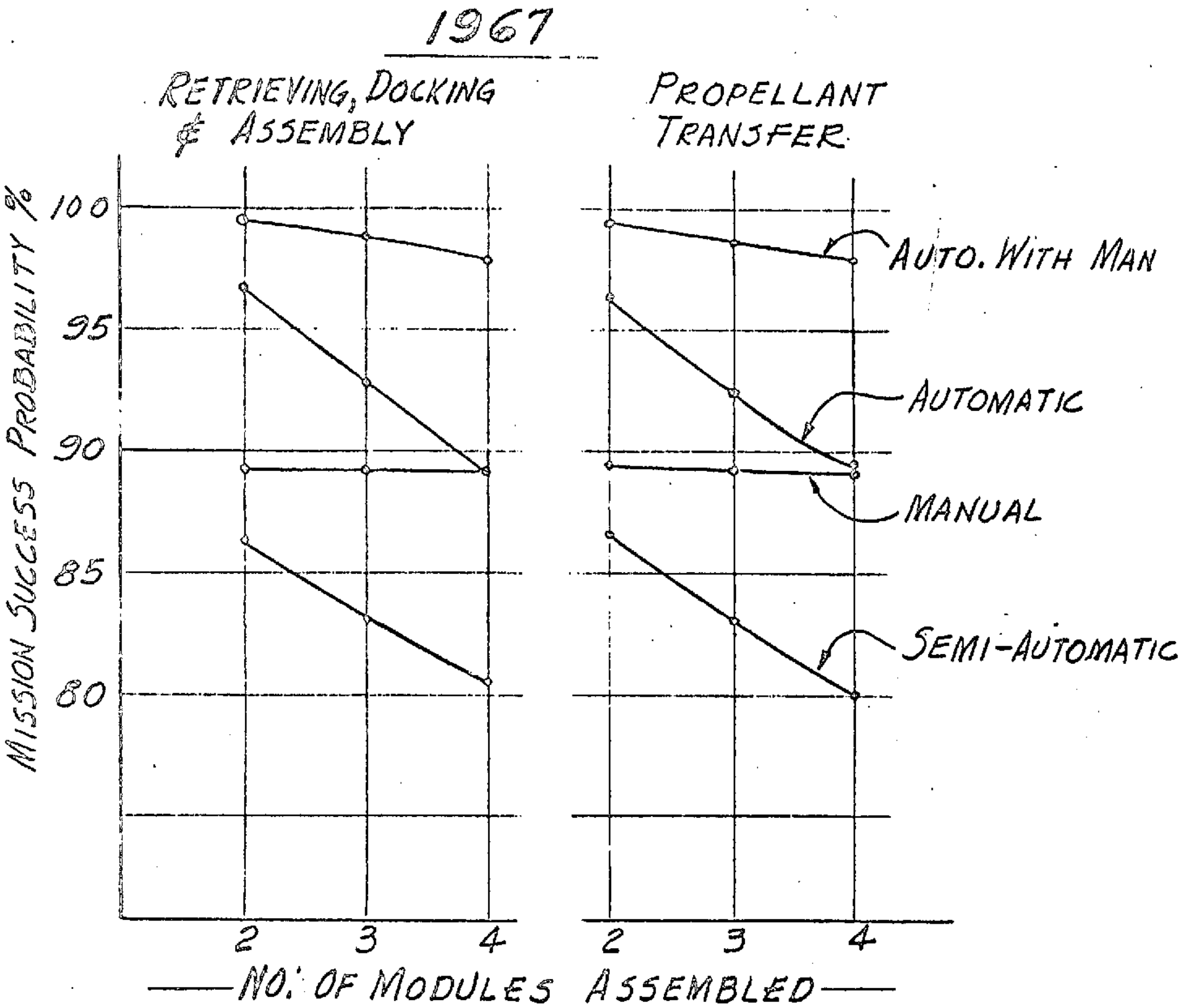
RETRIEVING, DOCKING & ASSEMBLY
RELIABILITY GROWTH CURVES

FIGURE 2-33



	1963	1964	1965	1966	1967
AUTO. WITH MAN	.9165	.9699	.9847	.9903	.9934
AUTOMATIC	.885	.912	.933	.950	.962
MANUAL	.4953	.7474	.8408	.8763	.8932
SEMI-AUTO.	.444	.688	.791	.836	.863

RETRIEVING, DOCKING, ASSEMBLY & FUELING
RELIABILITY GROWTH CURVES
FIGURE 2-34

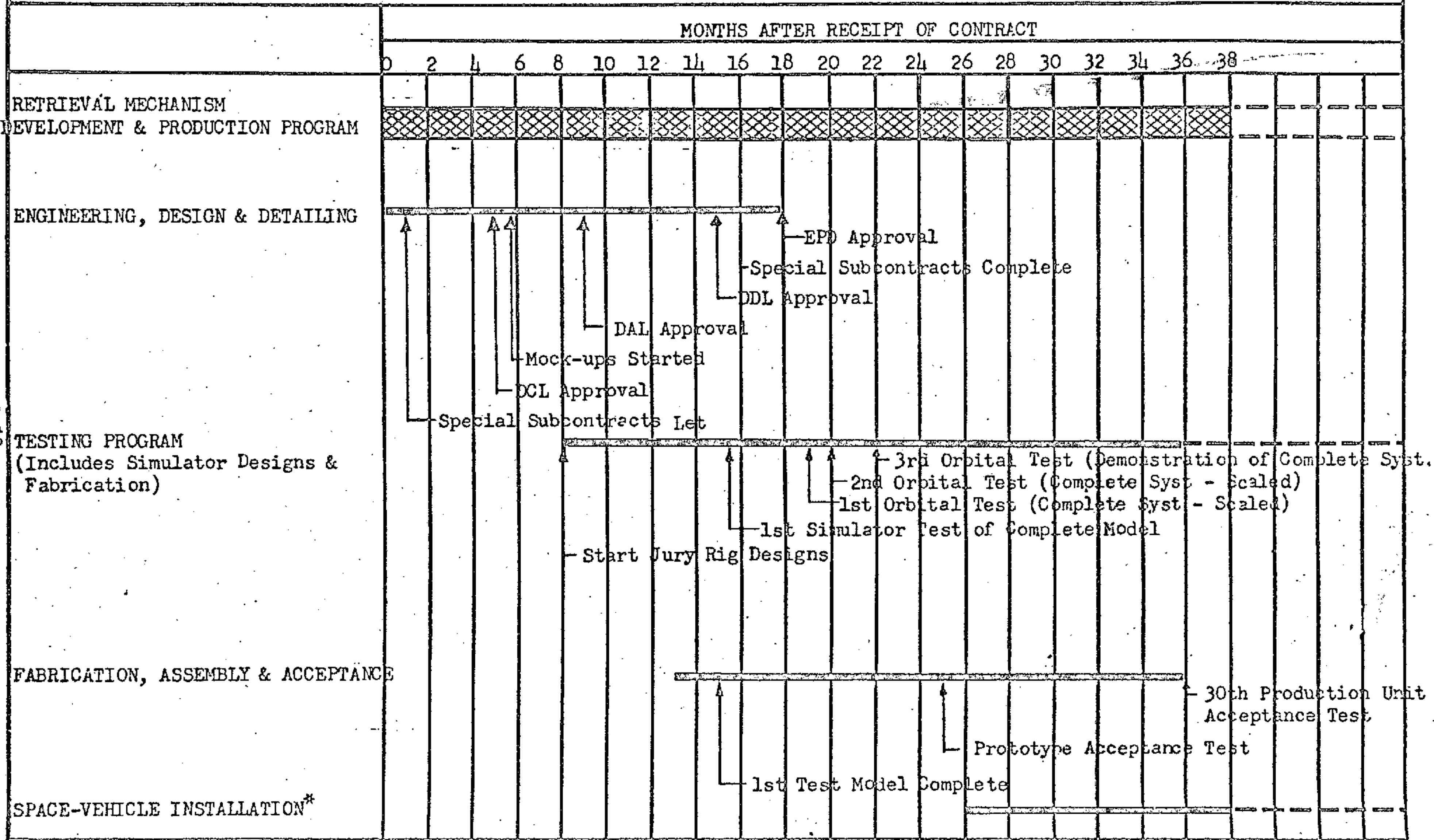


	No. OF MODULES ASSEM.	2	3	4
RETRIEVING DOCKING & ASSEMBLY	AUTO. WITH MAN	.994	.989	.9835
	AUTOMATIC	.963	.928	.893
	MANUAL	.894	.892	.891
	SEMI-AUTOMATIC	.864	.832	.804
ASSEMBLY WITH PROPELLANT TRANSFER	AUTO. WITH MAN	.9934	.9868	.9803
	AUTOMATIC	.962	.925	.891
	MANUAL	.8932	.892	.888
	SEMI-AUTOMATIC	.863	.83	.80

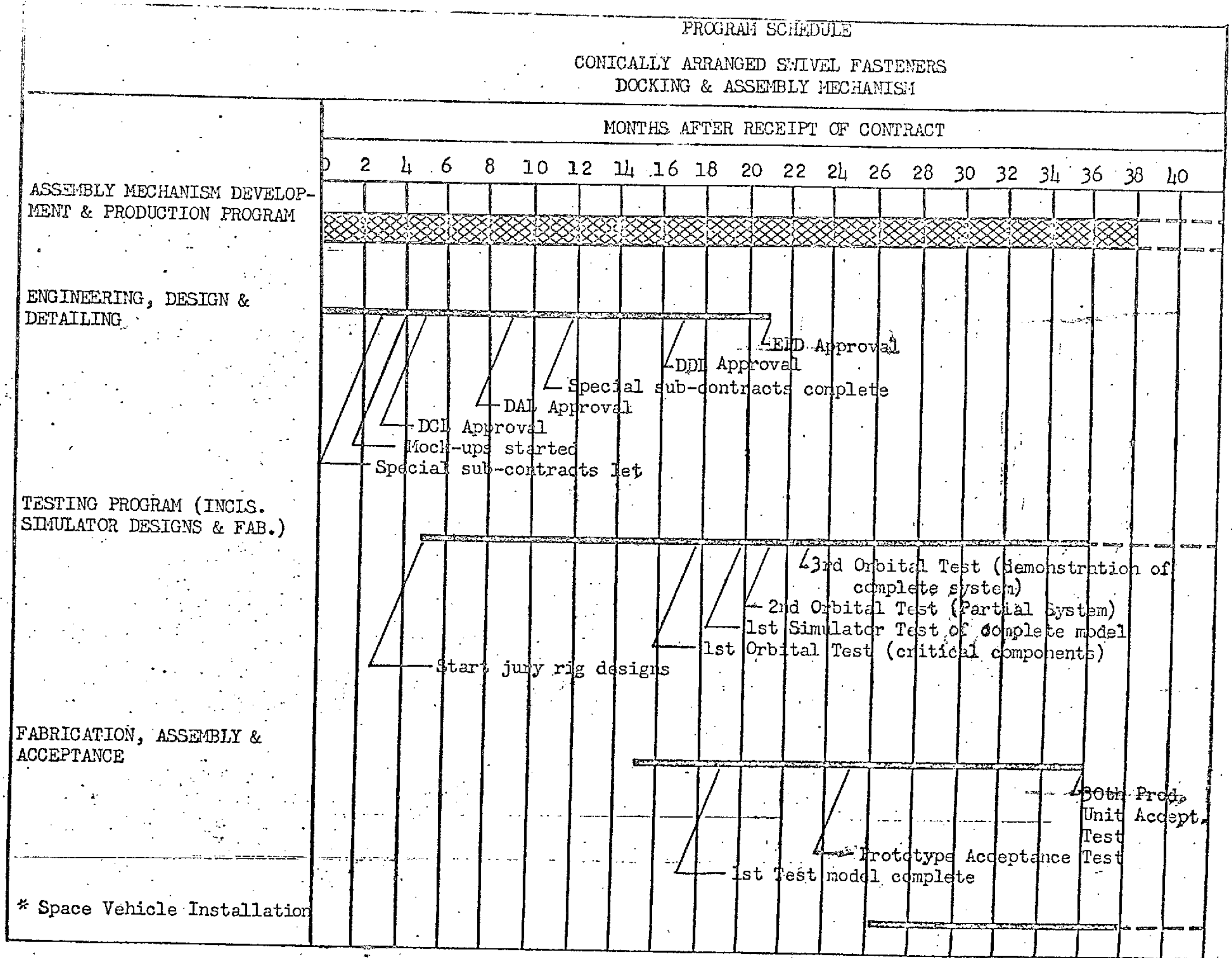
1967- RELIABILITY VS. NO. OF MODULES ASSEMBLED

FIGURE 2-35

PROGRAM SCHEDULE
RIGID BOOM - SIDE LOCATION
RETRIEVAL MECHANISMS

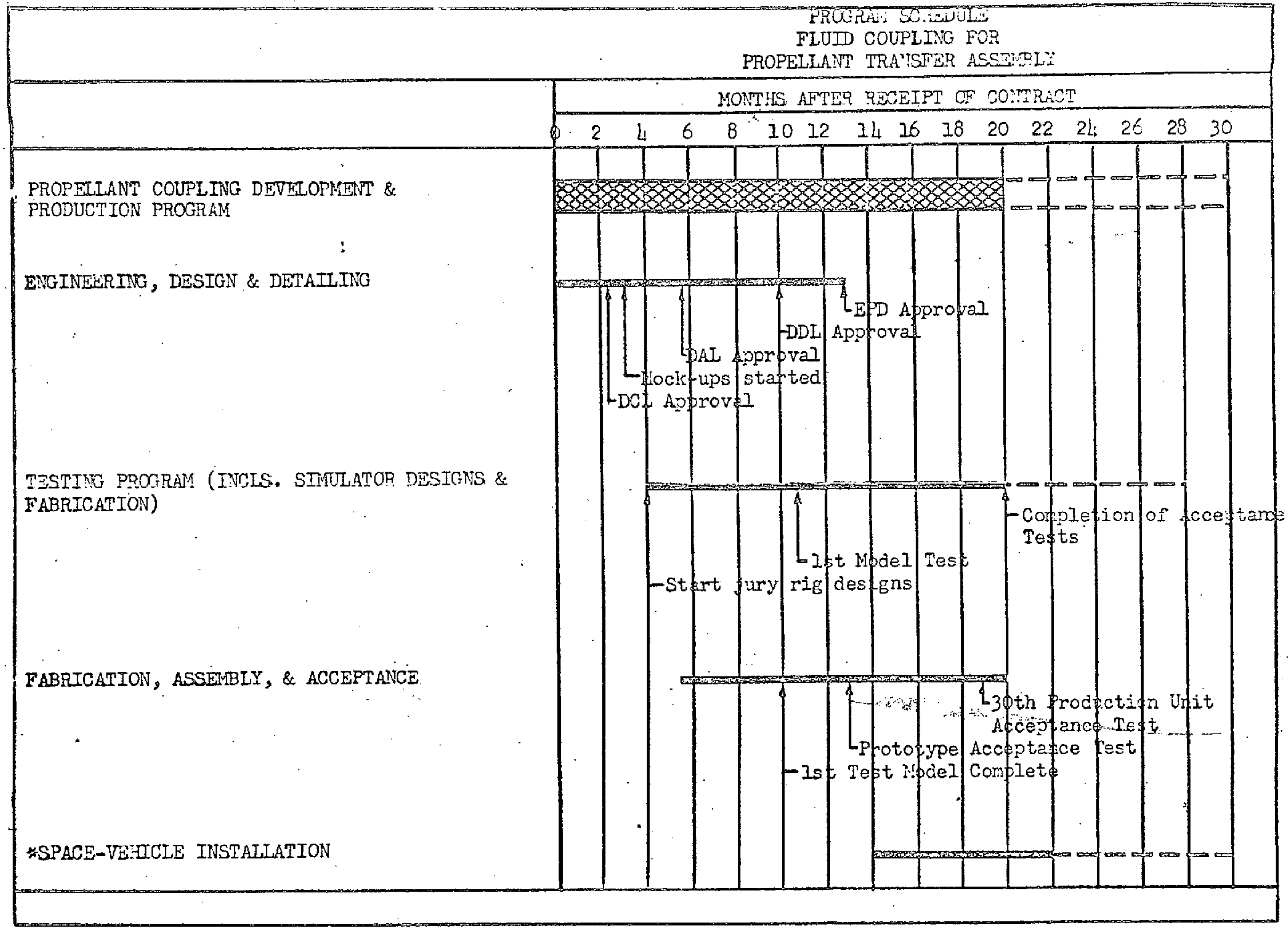


* For Specific Milestones, See Space Vehicle Development Program.



*For specific milestones see Space Vehicle Development Program

Figure 2.37



*For specific milestones see Space Vehicle Development Program

PART III
PROPELLANT TRANSFER

FOREWORD

Three major approaches of exploiting the orbital launch concept are orbital assembly, crew transfer, and propellant transfer. In support of the Orbital Launch Operations Study for NASA by the Chance Vought Corporation, Douglas Aircraft Company was requested to study and evaluate the concepts of orbital propellant transfer and to present a recommended concept for development. This report contains the results of that study.

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LIST OF ILLUSTRATIONS

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3.1 INTRODUCTION

3.1.1 General

This report presents design concepts of orbital propellant transfer methods capable of supporting lunar landing and circumlunar missions. The recommended propellant transfer concept is applicable to both types of missions. The report presents a general description and analysis of the tanker and orbital launch vehicles (based on existing or planned upper stages of Saturn) associated primarily with the lunar landing missions. However, specific design requirements applicable to the circumlunar mission are presented throughout the report. Propellant storage, docking maneuvers, tanker disposal and other necessary phases associated with orbital propellant transfer were briefly investigated; therefore, general conclusions, specifically related to these areas, were made to provide a basis from which to design the tanker vehicle.

One possible space vehicle, capable of lunar landing, consists of an Apollo type manned spacecraft, weighing approximately 175,000 lbs. and a Saturn S-II booster stage. To impart escape velocity to the spacecraft departing from a 300 nautical mile earth orbit requires approximately 260,000 lbs. of liquid oxygen-liquid hydrogen propellants. Since this is greater than the payload capability of earth launched Saturn vehicles, the propellant would be boosted into orbit in two tanker payloads and then transferred to the Orbital Launch Vehicle (S-II stage). The tanker vehicle, designated S-IVB, is in most respects, an enlarged Douglas Saturn S-IV upper stage vehicle. Each tanker has useable propellant capacity of 200,000 pounds of liquid oxygen-liquid hydrogen propellants.

A second space vehicle which could be considered for the lunar landing mission consists of an Apollo spacecraft weighing approximately 97,000 lbs. with a partially fueled (46,000 lbs.) S-IVB launch vehicle. In this case, approximately 122,000 pounds of propellant would be transferred from an S-IVB tanker vehicle to the OLV (S-IVB) prior to orbital launch.

A modified Apollo spacecraft weighing approximately 27,000 lbs, with an S-IVB launch vehicle stage might be used for circumlunar missions. Approximately 60,000 pounds of liquid oxygen-liquid hydrogen propellants would be transferred from three Saturn S-V tanker vehicles to the circumlunar OLV prior to orbital launch. Each tanker has a propellant capacity of 33,000 lbs. of propellant; however, only 91% of the propellant capacity of one tanker and 57% of the propellant capacity of the remaining two tankers would be required to fully load the OLV.

The S-IVB tanker vehicle proposed in this study does not contain a main propulsion system. The approach in this analysis was to assume that the tanker vehicle could be placed into orbit by an earth launch vehicle (booster). This method would not require propellant consumption from the tanker propellant payload. If additional impulse is required for the orbit, the tanker configuration is such that a propulsion system can be readily mounted. The proposed tanker makes maximum use of existing and currently tested components. Such a configuration will provide a highly reliable space tanker with minimum design and development time and effort.

In addition, during the course of this study, parametric data were generated for liquid oxygen-liquid hydrogen propellant transfer loads of 20,000, 100,000, and 200,000 pounds at a 5:1 mixture ratio. Whereas the data were developed in conjunction with the above described missions and associated hardware the results are general and are applicable to a family of similar launch vehicle and tanker configurations.

3.1.2 Study Criteria

The over-all OLO Study established certain objectives and guidelines for the study of propellant transfer.

3.1.2.1 Study Objectives

The objectives of the propellant transfer investigations were:

(a) Evaluate the various concepts of transfer of propellants under orbital conditions.
(b) On the basis of this evaluation, recommend a particular concept for development with the OLO program.
(c) Present a conceptual design of a system based on the recommended concept.
(d) Establish the development plan required to demonstrate the operational feasibility of the selected propellant transfer concept. The development plan should provide for the resolution of problem areas associated with this concept.
(e) Determine the reliability of the operation to support the over-all OLO mission success analysis.

3.1.2.2 Study Guidelines

In order to evaluate the concepts, the following guidelines were established:

(a) Primary investigation would be limited to LO_2-LH_2 propellant systems. "Storable" propellant systems would be investigated only to the extent necessary to establish how they differ from the LO_2-LH_2 systems.
(b) The OLO missions are circumlunar, lunar orbit, and initial manned lunar landing.
(c) The orbital propellant transfer investigations are to be performed for two conditions:

(1) Including an OLF (Orbital Launch Facility)
(2) Excluding an OLF

(Primary emphasis is to be placed on the latter condition only because of insufficient time to perform both investigations.)
(d) Propellants are to be transferred:

(1) From propellant supply vehicles (S-V, S-IV, S-IVB, S-II) or a new tanker stage.
(2) To a family of OLV's and spacecraft having escape payloads of 30,000 to 180,000 pounds.

(e) The transfer concepts to be considered are:

(1) Linear Acceleration
(2) Rotating Assembly
(3) Bladders
(4) Pistons
(5) Momentum Transfer
(6) Supercritical Storage

(f) Storage concepts will be considered with regard to the effect on the tanker design and transfer operation. (A nominal storage period of 30 days was considered.)

(g) Criteria for selection are:

(1) Feasibility
(2) Reliability
(3) Availability and Costs
(4) Maintainability
(5) Physical Characteristics - weight, volume, power requirements
(6) Time in Orbit

(h) Consider orbital altitude of 450 - 600 KM (556 KM or 300 nautical miles was considered as representative).

(i) The terminal docking conditions are:

Range	0.5 ± 0.5 feet
Range Rate	0 ± 0.01 fps
Lateral Displacement	0 ± 1.0 feet
Lateral Velocity	0 ± 0.01 fps
Angular Alignment	0 ± 1°
Angular Velocity	0 ± 0.1°/sec

(j) Misalignment tolerances after docking are:

±4° angular
±0.12 inches assembly

(k) Manpower available for transfer operations

(1) Without OLF, three Apollo crewmen
(2) With OLF, three Apollo crewmen plus two OLF crewmen external, and two OLF crewmen internal.

3.2 RECOMMENDATIONS

The various transfer concepts (see (e) above) were evaluated with respect to the established criteria and the linear acceleration system is recommended for development for early orbital launch operations. A primary factor in the evaluation was that this system was considered within the present state-of-the-art. For advanced operations, rotating assembly and bladder systems should also be considered, although the linear acceleration system appears to maintain a distinct advantage. The presence of an orbital launch facility increases the interest in, and feasibility of, the rotating assembly and bladder systems; however, the linear acceleration concept would also be compatible with an OLF.

A research program should begin at the earliest possible time to investigate the behavior of LO_2 and LH_2 under zero and micro g conditions. Present analysis indicates that an acceleration of 0.001 g for ten minutes is ample for the transfer of 200,000 pounds of LO_2-LH_2 (at 5:1 ratio) with a 2,000 pound pressurization system. The reliability of the transfer operation itself is estimated as 0.94 for the latter part of 1965 and 0.976 for the 1967 time period.

3.3 DISCUSSION OF CONCEPTS

3.3.1 Description

The propellant transfer systems considered are conceptually illustrated in Figure 3-1. The tanker is boosted into orbit where the space vehicle attaches to it by an end docking method. The propellant is then transferred, the tanker detached, and removed, and the space vehicle begins its checkout and countdown procedures. In some cases where more than one tanker is required, the space vehicle docks and fuels from each in turn. The systems considered were:

(a) Linear Acceleration
(b) Rotating Assembly
(c) Bladders
(d) Pistons
(e) Momentum Transfer
(f) Supercritical Storage

All but the last, supercritical storage, are illustrated in Figure 3-1.

3.3.1.1 Linear Acceleration

The linear acceleration system imposes a settling acceleration on the propellants to achieve a vapor-liquid separation to collect and maintain the liquid at the outlet sump. Either a pump or tank pressurization system is then employed to force the propellant out of the tanker and into the OLV.

Analytical studies conducted at Douglas indicate that accelerations on the order of 10^{-5} g are sufficient for settling and maintaining liquid-vapor separation in liquid hydrogen. For conservative reasons, based on lack of empirical support, a higher acceleration level is desirable. An acceleration on the order of 0.01 g or greater requires an inordinate amount of propellant. Based on these considerations an acceleration of 0.001 g was selected for the system. For the pressurization system selected, the transfer time during which acceleration must be maintained is 10 minutes. Thus for an assembly of 400,000 pounds total weight, a total impulse of 240,000 pound-seconds is required during transfer. For a storable hypergolic propulsion system, total acceleration propellant weight is about 850 pounds and total acceleration system weight about 1,100 pounds. Additional propellant can

TRANSFER SYSTEMS CONCEPTS

ARTIFICIAL "G" SYSTEMS

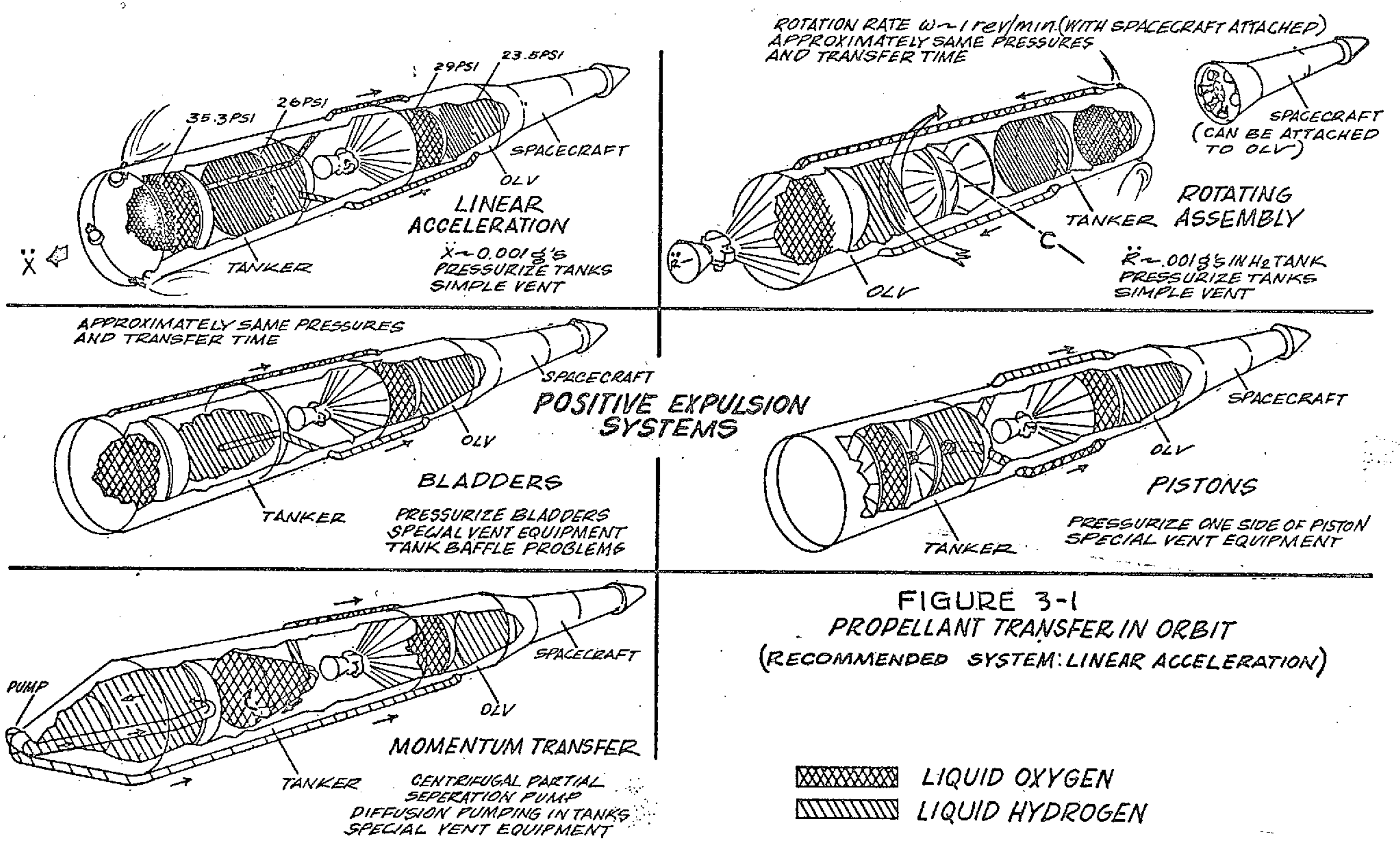


FIGURE 3-1
PROPELLANT TRANSFER IN ORBIT
(RECOMMENDED SYSTEM: LINEAR ACCELERATION)

be provided for orbit adjustment, and for ullage to vent during orbital storage. The linear acceleration concept lends itself to a straightforward solution to the venting problem in a zero g environment (after phase separation, normal venting is effected).

The system shown in Figure 3-1 indicates a "rearward" acceleration of the assembly during transfer. This is done for two reasons: in this direction the acceleration aids rather than hinders the transfer process, i.e., it is "downhill" from tanker to receiver, and the "bottom" of the tanks are closer to the OLV, allowing shorter transfer lines. This is important in that lower pressure differentials are required for transfer in a given time or conversely, for a given maximum pressure, shorter transfer time is required.

The suggested hookups, i.e., with the docking face of the tanker on the hydrogen tank (upper end when the tanker is boosted from earth), is selected so that the tanker could still be fitted with a main propulsion system for injecting into orbit if desired. This would require some other compromises with the tanker design, but may prove necessary.

3.3.1.2 Rotating Assembly

The rotating assembly system also imposes a settling acceleration of the propellants to achieve vapor-liquid separation and collect and maintain the liquid at the outlet sump. Either a pump or tank pressurization system is then employed to force the propellant out of the tanker and into the OLV.

As in the case of linear acceleration, an acceleration of approximately 0.001 g is desired in the propellant tanks. In a rotation system, however, the acceleration increases linearly with the distance from the center of rotation. Since the assembly will rotate about it center of gravity, the distribution of masses and vehicle configuration is an important consideration in the propellant transfer operation. Figure 3-1 shows one concept of a rotating assembly propellant transfer, with the manned spacecraft detached from the rotating assembly. This would be the case only if high rotations ($\omega > 4$ rpm) were required to produce the necessary accelerations. For the case where the spacecraft is attached to the OLV and the tanker is at the rear of the OLV, the c.g. will initially be in the area of the docking interface between the OLV and tanker and approximately 170 feet aft of the manned capsule. Imparting a rotation about the pitch axis of $\omega = 0.43$ rpm will give an acceleration of 0.001 g's at 15 feet, which is in the upper region of the hydrogen tank. As the propellant is transferred, the c.g. will shift forward and the spin rate will automatically increase to a maximum of about 1.3 rpm. Preliminary studies indicate that attitude control is not required for this system with the described configuration. According to information on human tolerances, spin rates of 4 rpm are acceptable for the manned vehicle and therefore the spacecraft need not be detached; in fact, the presence of the spacecraft tends to counterbalance the tanker and improve the dynamic stability of the vehicle by increasing the moment of inertia of the total assembly and by shifting the c.g. forward from the tanker. It also provides a spin up and decelerating system as well as an attitude control system if needed.

Other rotation systems may be envisioned where only the tanker is spun up and the propellant is transferred to a stationary OLV and spacecraft. This system appears to offer no advantages over the rotating assembly except allowing the manned capsule to remain stationary if high spin rates are required; however, as previously mentioned, this is not the case. On the other hand, a rotating tanker system has several disadvantages. Propellant must be transferred through a rotating joint which is heavy and complicated. The tanker will be rotating about a c.g. located somewhere in its tank area, requiring higher spin rates and distributing the propellant "around" the tank rather than at the "bottom". It will impart some motion to the OLV through the joint and through the propellant transfer, thus requiring attitude control of that craft during transfer if it desires to remain stationary. Because of the greater relative inertia change, the spinning tanker will tend to rapidly increase its spin rate by a much higher factor than three times the initial, as was the case for the rotating assembly. For this reason the tanker will probably require an attitude control and decelerating system as well as a spin up system.

The rotating system lends itself to straightforward venting during transfer and storage.

In systems where large amounts of propellant are stored in orbit, constantly supplied from earth launched tankers, the orbital filling station concept, i.e., a large slowly rotating "fuel dump", offers certain advantages. For early missions, however, where each OLV has special propellant tanker(s) assigned to it, linear acceleration appears to offer a simpler and more straightforward method of propellant transfer at the same, or nearly the same, weight and power requirements.

3.3.1.3 Bladders

As opposed to the simulated "g" systems previously discussed, the propellants may be moved in total by mechanical action and thus forced out of the tanker and into the OLV. One such positive expulsion concept would utilize thin flexible bags, or bladders, either enclosing the propellant or in a position so that when expanded the bladder will fill the confines of the tank, forcing the propellant out. Figure 3-1 illustrates a bladder system. Studies to date indicate that a bladder is required only in the tanker, and that the receiver is pressurized to slightly higher than the propellants vapor pressure to prevent excessive boil-off as the propellants enter.

The vent system, however, is complicated by the fact that no definite liquid-vapor separation occurs in the positive expulsion system and there is no assurance that vapor, and not liquid, will be at the vent orifice. Since the venting of liquid overboard could lead to prohibitive propellant losses, some means of separation must occur in the venting system. This requires special vent equipment, such as the Janitrol centrifugal vent assembly. Investigation of this equipment, which was designed for the Centaur vehicle, has determined that it is not capable of handling the gas flow rates associated with the transfer operation. The design and concept are excellent, although the increased requirements (e.g., 15 to 20 times larger for a 200,000 lb. OLV loading) may prove a serious weight penalty.

Although considerable work has been done in developing a suitable bladder material for cryogenic propulsion systems, many problems remain, especially in developing a very large bladder such as would be required for propellant transfer. The properties of bladders under long time storage of cryogenics also need further study, especially with respect to the possibility of osmosis of the propellant through the bladder and the deterioration of the bladder material.

The compatibility of bladder systems with existing tank structure and equipment storage techniques, such as the placement of helium bottles in the hydrogen tank, must be considered. Since the bladder must have unobstructed free access to all parts of the tank interior to purge the propellant, equipment mounted in the tank must be either moveable or shaped to allow the bladder to conform to its surface. Some particular installation problems involve the baffles required in the tank to prevent sloshing, the propellant lines, the vent system, and the sensing probes mounted in the tank to take propellant measurements. The tank structure itself must be considered. For instance, an internal waffle tank skin is not likely to be compatible with a bladder. Unless great care is taken in the design and development of the bladder, e.g., the expansion (or contraction) rate in its various sections, etc., the bladder itself may "trap" propellant in the tanker. Also, considerable propellant may be trapped in the transfer lines since the bladder cannot purge these.

With respect to system weight, the bladder system requires a heavier pressurization system since the helium heater is not compatible with non-ullage systems as it requires positive propellant flow at a stated quantity. An auxilliary stored gas pressurization system could be devised such that it would initiate propellant expulsion from the bladder, which would then provide a liquid propellant flow to the helium heater. The helium heater could then "bootstrap" propellants with heated helium gas pressurizing the bladder and complete the propellant transfer. This additional weight plus the weight of the bladder itself should be compared to the propulsion system employed in linear acceleration system.

3.3.1.4 Pistons

Pistons are another positive expulsion transfer concept similar to bladders in many ways, except that a diaphragm is translated the length of the propellant tank instead of expanding (or contracting) throughout the tank. The structural problems appear more severe but material problems appear less than for the bladders. System weight is also probably higher although this requires further investigation.

The piston would tend to suppress sloshing; however, it would have to resist the dynamic forces of the liquid propellants. In addition, the clean tank interior problem would be amplified and the obvious sealing problem would exist. It is probable, however, that leakage past the piston would not be great as there would not be a large pressure differential across the piston diaphragm.

3.3.1.5 Momentum Transfer

The momentum transfer system is conceptually illustrated in Figure 3-1. A centrifugal pump picks up whatever is available at one end of the tank, separates some of the liquid and transfers it, cycling the remaining fluid in a spray from the opposite end of the tank, thereby transferring the spray momentum to the remaining propellant to "wash it down" to the vicinity of the pump. There will be some residual propellant remaining in the tanker after transfer since there is some minimum amount of fluid required for the spray. The amount cannot be determined at this time, but it may be considerable. As indicated in the sketch, the system may be sensitive to tank shape and size . This system inherently utilizes pumps for transfer and thus requires an adequate power source for their operation. This may be supplied during the transfer operation by power sources on-board the spacecraft. The transfer time for this system will be a function of the pumps, the amount of propellant, the pressure losses in the lines, and efficiency of the momentum transfer operation. It cannot be determined at this time, but may considerably exceed the 10 minutes required by the other systems.

3.3.1.6 Supercritical Storage

This method of propellant transfer stores the propellant at a high enough pressure and temperature to maintain all the propellant in the form of a one state fluid which is expanded in the transfer process to arrive in the OLV as liquid. The use of supercritical hydrogen as a means of propellant transfer in orbit does not appear practical based on the available information about this method. The hydrogen temperature entropy chart shows that it would be marginal if not impossible to throttle from a supercritical state to a liquid condition without entering the two-phase region which is not desired. This plus localized effects points up the undesirability of this method of transfer.

3.3.2 Problems of Propellant Transfer Under Orbital Conditions

3.3.2.1 Zero "g" Environment

The foremost problem associated with transfer of propellants under orbital conditions is the zero "g" environment. The effect is that the liquid and vapor portions of the propellant are likely to be intermingled within the tank and the liquid cannot be expected to flow "downhill" into the pump intake or to be forced into an outlet by tank pressurization.

Several concepts for solving this problem present themselves. The simplest solution is to circumvent the problem by providing an acceleration to the tank to settle the propellant (separating the liquid-vapor phases) and force the liquid to the "bottom" of the tank, then the acceleration is maintained during the transfer operation which may be performed by pumping or tank pressurization. Another method utilizes a bladder which envelopes the propellant (or bounds it against the tank walls) and, by pressurizing one side, the bladder contracts (or expands) to expell the propellant, both vapor and liquid, out of the tank.

3.3.2.2 Vacuum

For all practical considerations of propellant transfer, a high vacuum may be assumed at the orbital altitudes considered. This condition has a mixed effect upon the operation. On the one hand, sealing problems may be more severe than in the atmosphere and leaks more likely to occur. Also, the phenomenon of "cold welding", or the bonding of surfaces brought into contact in vacuums can lead to many mechanical problems, in the docking or coupling operation, for example. Proper design, material research, and lubrication methods may alleviate these problems.

On the other hand, a high vacuum environment offers some advantages over an atmospheric environment for propellant transfer operations. If leaks are more prone to occur, it is also true that they are likely to prove less dangerous. The dispersion of the gases under vacuum conditions will be so rapid and complete that combustible concentrations are very unlikely to occur. In addition, there is no oxidizing element present and therefore, a leakage of both oxidizer and fuel must occur to provide the possibility of combustion. Under atmospheric conditions, frost and ice formation may occur on equipment containing cryogenic propellants. This can lead to mechanical and electrical problems and in some cases require the use of a purging system. This should not manifest itself in the orbital environment, although, if the propellant leakage or venting is extensive, it is a possibility.

3.3.2.3 Thermal

The thermal environment in orbit affects the propellant transfer study primarily in its effects upon orbital storage of the propellants and the necessary tie-in of any propellant transfer concept with the storage system. The requirements of tank structure, insulation, venting operation, and propellant monitoring dictated by the storage concept may be mitigated or increased by the transfer concept selected. Similarly, the storage system may favorably or adversely affect the transfer system.

In any storage concept allowing boil-off, some method of venting vapor from the tanks is required. Thus a liquid-vapor separation technique or equipment is required. In the use where transfer utilizes an acceleration concept, for instance, the same solution may be applied to the venting. Whenever venting is required during storage, the propellant may be settled for the few seconds required for venting. For other systems, such as bladders, some type of separating device may be utilized as an acceleration system incorporated exclusively for venting purposes.

Another aspect of storage, with LO_2-LH_2 systems, is the problem of preventing the cold hydrogen from freezing the LO_2. This requires more isolation and insulation between the two tanks than is usually employed for LO_2-LH_2 booster systems.

The thermal environment presented by the cryogenic propellants themselves must be considered in the design and construction of the propellant couplings, bladders, etc., associated with the transfer system.

3.3.2.4 Radiation

The low altitude associated with orbital operations should preclude any problems associated with radiation in space. However, the effect of prolonged exposure of the materials (during orbital storage) to the low altitude radiation should not be dismissed without further investigation.

3.3.3 Discussion and Evaluation

3.3.3.1 General State of Art

The results of recent propellant-transfer studies, by various organizations in the space industry, have indicated that this operation is firmly within the realm of present technology and component design. The emphasis in this current study has been to further assure that complete feasibility is possible by analyzing and designing a detailed hardware system which can accomplish the transfer mission. Since a primary requirement in support of a detailed system design is the availability of hardware components and subcomponents, contacts were made with the appropriate industries to obtain design information. The principal components surveyed were available control valves, quick disconnects, in-flight drogue and probes and related hardware.

The results of this survey, definitely prove that hardware is becoming available in the smaller sizes and with the design details that will be required. Several of the supporting industries (e.g., CALMEC Mfg. Co., Fairchild Stratos Corp.) are in the process of developing suitable cryogenic propellant components.

The analyses conducted in this study were as comprehensive as necessary to provide adequate proof that vehicle systems and subsystems are capable of meeting the necessary propellant transfer requirements. These are described in detail in the following sections. The necessity of conducting sufficient trade-off studies to reach a reasonable optimization of system and counter-system is evident and the critical trade-off studies have been conducted. These included, for example, an investigation of propellant transfer time as a function of ullage propellant weight, propellant transfer time as a function of propellant line size and weight, weight comparison of pressurization systems required to achieve propellant transfer, et cetera. The conclusions reached, following this preliminary design investigation, is that orbital propellant transfer is feasible and is well within the scope of current system and funding capability.

When components were not available, modifications to existing hardware as well as preliminary designs were investigated. The most frequent characteristic lacking in available components and in current on-the-board designs is sufficient size. Imaginative design coupled with sound reliability features was evident in many components which were reviewed.

3.3.3.2 Associated Problem Areas

The investigations conducted to date on methods of orbital propellant transfer may be categorized as advanced design studies. This applies to the work accomplished within the Douglas Aircraft Company and of the reports which have been reviewed in the literary search phase of this study. The advanced design phase of a procedure as involved and unknown as "orbital transfer" is a necessary phase. However, while all of the initial studies proved on the basis of fundamental physics that transfer can be accomplished with reasonable vehicle weights and system complexities, it became apparent that effort directed toward the preliminary hardware design phase was necessary. Therefore, this study was directed at providing both a conceptual and preliminary hardware design for the purpose of uncovering unforeseen problem areas as well as more closely defining the foreseen and previously established problem areas. To properly evaluate a system's capability for accomplishing unproven tasks the vehicle designer must evolve firm hardware system and component design. It is only when this has been accomplished that it is permissible to categorically state the possibility of successfully accomplishing the mission under study. Examples of unforeseen problem areas encountered, when studying the orbital operations required to achieve vehicle rendezvous, useful propellant transfer, and subsequent vehicle mission are: efficient gas-liquid-droplet separation during venting under high propellant flow rates and low vehicle longitudinal accelerations, propellant gaging instrumentation and a safe reliable method for removing the docking structure and unnecessary propellant transfer lines from the receiving vehicle, heat leaks through propellant tank fittings (e.g., main propellant feed line tank fitting), and provisions for a clean and uncluttered propellant tank interior for use with positive expulsion bladders.

3.3.3.3 Criteria Evaluation of Transfer Concepts

In order to evaluate the transfer concepts and rank them in order of feasibility, reliability, and maintainability, an analysis was conducted in which four designers, each familiar with the six systems considered, were requested to give pairwise system preferences for various criteria. Numerical analyses were then performed on these paired ratings and the result was a ranking of the systems which shows not only the actual order but the relative magnitude between rankings.

The feasibility criteria were selected from the reliability criteria and modified by the multiplication by weighting factors (W.F.). The feasibility criteria and their weighting factors were:

	W. F.	Criteria
(a)	1	Complexity of venting system
(b)	1	Operating time
(c)	3	Functional Complexity of system
(d)	2	Environmental effect
(e)	4	State of system development
(f)	1	Flow rate control
(g)	3	Dynamics problems
(h)	1	Human control provided

The reliability criteria included all the criteria for feasibility plus the following:

(a) - (h) Above
(i) Liquid-vapor separation
(j) System physical complexity
(k) Necessity for auxiliary equipment
(l) Maintainability

Maintainability was evaluated separately as well as a part of the reliability. The results of these ratings are presented in Table 3-1.

In addition to this evaluation, the additional system evaluation criteria were rated according to preliminary studies of each concept where results were indicated. This rating is also included in Table 3-1. When several concepts have the same number it indicates that they were ranked equal relative to each other; where numbers appear between concepts, it indicates the relative variation in ranking.

TABLE 3-1
EVALUATION OF TRANSFER CONCEPTS

Feasibility	Reliability	Availability and Costs	Weight Volume & Power Require.	Maintain-ability
1. LA	1. LA	1. LA	1. LA, RA	1. LA,RA,P
2. -	2. -	2. B, RA	2. B	2. B
3. -	3. -	3. P	3. P	3. MT
4. B, P	4. B, P	4.	4. MT	4. SS
5. RA, SS	5. RA, SS	5. MT, SS	5. SS	
6. MT	6. MT			

Abbreviations: L.A. - Linear Acceleration
B. - Bladder
R.A. - Rotating Assembly
P. - Pistons
M.T. - Momentum Transfer
S.S. - Supercritical Storage

3.3.3.4 Propellant Transfer System Evaluation

A sound orbital propellant transfer design philosophy should produce a system based on the following criteria:

(a) An operational principle embodying simplicity and practicality.
(b) Reliability
(c) Economy
(d) Design simplicity in system and component
(e) Minimum number of required operational manipulations

The linear acceleration method of orbital propellant transfer (a combined system of low acceleration with mass transfer by pressure) satisfactorily fulfills the above stated requirements and surpasses other feasible design concepts. The advantages and disadvantages of this method and areas requiring further study are summarized below.

(a) Advantages

(1) The operating principle and necessary supporting systems are basic in concept and design.

(2) No moving or rotating components are necessary; therefore, compensation is not required to balance potential reaction forces.

(3) Propellant transfer can be achieved in a reasonable time.

(4) The separation of propellant liquid-vapor phases prior to transfer in a zero "g" environment can be accomplished with a proven method of linear acceleration.

(5) Propellant transfer can be performed with a proven, highly reliable (pressure) technique.

(6) The selected low acceleration level (10^{-3} g's) for propellant liquid vapor phase separation is below the critical value that would cause propellant sloshing.

(7) System rotational deviations caused by unbalanced propellant flow momentum are negligible due to the high moment of inertia of the system and therefore does not require attitude control corrections.

(b) Disadvantages

(1) A weight penalty is incurred as a result of the auxiliary propulsion subsystem required to accomplish phase separation.

(2) The auxiliary propulsion subsystem requires redundancy (two motors on each side) to increase system reliability.

(3) A slight change occurs in the mission orbital trajectory due to the induced velocity increment (e.g., 18 ft/sec) resulting from acceleration of the space vehicle and tanker vehicles. Consequently, a new flight trajectory is required for the new space vehicle position.

(c) Areas Requiring Further Study

(1) Liquid and vapor behavior in zero "g" environment

(a) When stabilized (V = constant)
(b) When agitated due to small perturbations
(c) For a range of low vehicle accelerations to establish the minimum acceptable accelerations

(2) Same conditions as (1) above with heat inputs (solar plus albedo radiation) to define more accurately heat transfer characteristics (change in saturation pressure as a function of propellant bulk temperature rise, boil-off rate, etc.).

(3) Dynamic disturbances (impulse, angular momentun, coriolis acceleration, etc.), of a complex system in an equilibrium zero "g" condition.

3.4 RECOMMENDED PROPELLANT TRANSFER SYSTEM

3.4.1 System Description

The propellant transfer system proposed in this study is a combination of two separate subsystems:

(a) A low thrust auxiliary hypergolic or cryogenic propulsion subsystem which produces linear acceleration for propellant phase separation, and

(b) A propellant tank pressurization subsystem with all the necessary components and hardware required to accomplish propellant transfer.

A pressurization system, rather than a pumping system, was selected as the prime method for effecting the propellant transfer because of the following factors:

(a) A pressurization system is required to suppress cryogenic propellant boil-off and meet Net Positive Suction Head (NPSH) requirements of the main propulsion system (if employed)

(b) A propellant transfer pump would require propellant tank pressure to meet the NPSH at the transfer flow rates and low artificial gravity forces

(c) The propellant transfer pump requires an energy source and adds mechanical complexity to the transfer system

(d) A pump would impart rotational torques to the space assembly and thus complicate and add additional attitude control requirements.

3.4.1.1 Auxiliary Propulsion Subsystem

The auxiliary propulsion subsystem consists of two separate and independent plug-in type power packs. Each power pack unit contains a throttlable, hypergolic, pressure-fed engine which is designed to deliver 225 pounds of thrust. A pair of engines are externally mounted on each assembly panel as shown in Figure 3-2. Only one of these engines is used during transfer system operations. The additional engine is installed for reliability. In the event of failure, the other engine would automatically ignite thereby assuring complete orbital propellant transfer. The storable propellant (e.g., IRFNA and MMH) and helium pressure spheres are installed as a compact unit assembly behind the mounting panel. This unit weighs approximately 560 pounds and is designed so that the engines can be rotated 180°. A schematic diagram of the auxiliary propulsion subsystem is shown in Figure 3-3. The components employed in this propulsion subsystem concept are readily available from existing hardware, thus assuring a reliable and economical design.

3.4.1.2 Tank Geometry

A general configuration of the orbital tanker vehicle propellant tank is shown in Figure 3-4. Although this diagram shows the dimensions for a propellant transfer capability of 200,000 pounds (oxidizer-to-fuel mixture ratio of 5:1), the over-all configuration is applicable to the other propellant transfer loads. Pertinent sizing parameters for the 200,000 pound

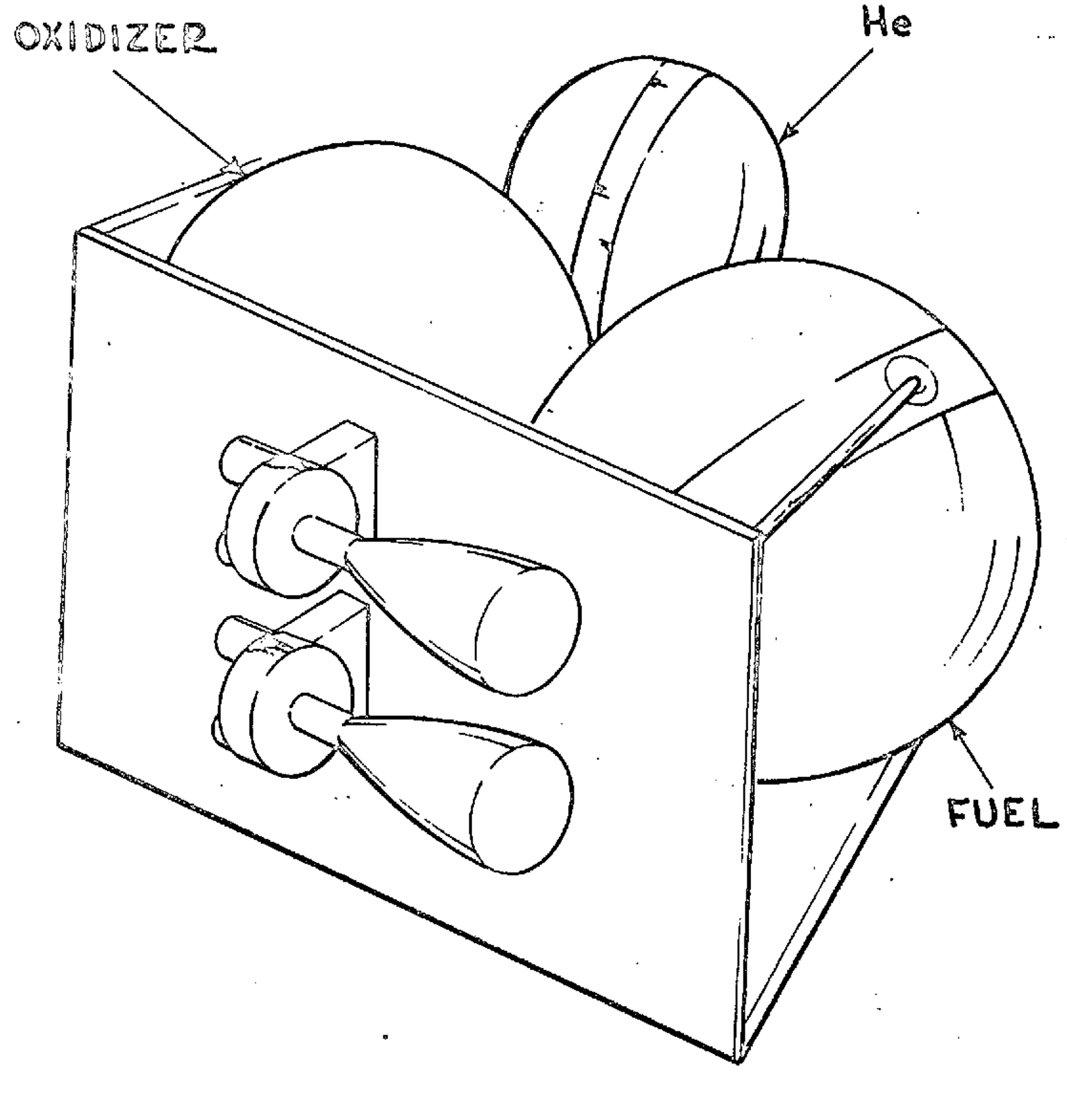


FIGURE 3-2

AUXILIARY PROPULSION SUBSYSTEM SCHEMATIC

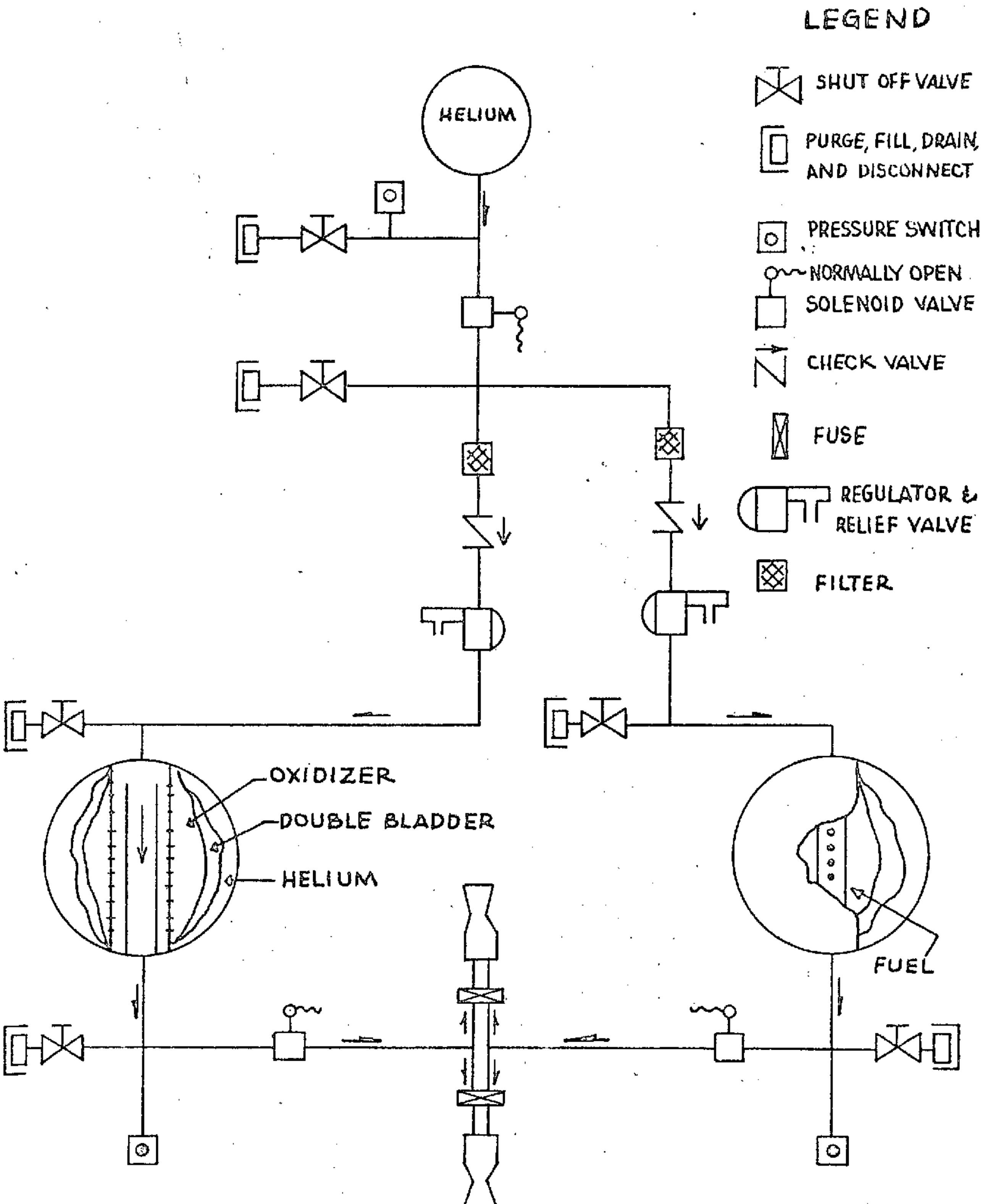
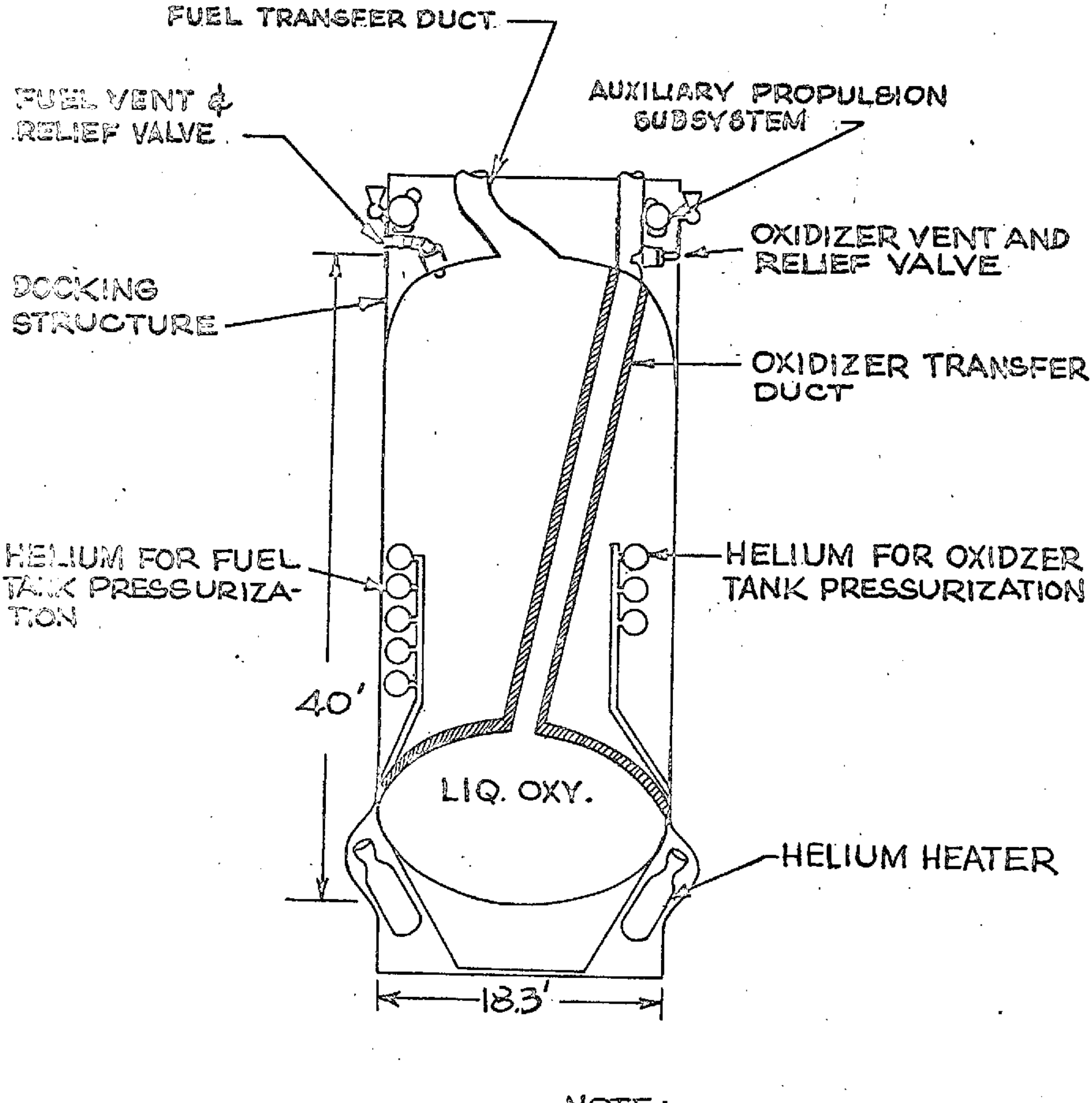


FIGURE 3-3

TANKER VEHICLE CONCEPT
SHOWING MAJOR SUBSYSTEMS



NOTE:
NOT DRAWN TO SCALE

FIGURE 3-4

capacity propellant tanks are listed below. For this particular mission the propellant tanks for both the tanker and the orbital launch vehicles were assumed to be identical.

	Fuel (LH_2)	Oxidizer (LO_2)
Propellant Weight (Lb.)	34,333	166,667
Total Tank Volume (Cu.Ft.)	8,148	2,489
Propellant Volume (Cu.Ft.)	7,822	2,389
Tank Ullage Volume (Cu.Ft.)	326	100

3.4.1.3 Pressurization System

The tanker vehicle pressurization system consists of two heat exchangers, cold helium storage vessels, and plumbing necessary for satisfactory operation. As shown in Figure 3-5, the helium is stored at 3,000 psia and at 50° R within a battery of spherical containers which are immersed within the liquid hydrogen tank. The bottles are manifolded together and all high pressure fittings are external to the hydrogen tank. During normal operation, helium gas flows through the regulator, which maintains a constant downstream pressure of 500 psia. The orifices downstream of the regulator further reduce the helium pressure before it enters the heat exchanger which heats the helium to 200°R. The two helium heaters used for propellant transfer are mounted 180° apart for vehicle dynamic stability. Each heater exhaust jet develops about 25 pounds of thrust; therefore, the heaters are positioned so that their thrust may be utilized as a supplementary power source for linear acceleration. For comparison, the pressurization and vent systems of the orbital launch vehicle booster are shown in Figure 3-6.

3.4.1.4 Propellant Transfer Ducts

A plan view of the tanker vehicle propellant transfer ducts is shown in Figure 3-7. The fuel and oxidizer transfer line outlet diameters are ten inches and eight inches, respectively. The liquid hydrogen transfer line is 31 feet long; the liquid oxygen transfer line is 56 feet long. Both propellant transfer line inlet diameters of the orbital launch vehicle booster are the same as the respective tanker vehicle duct outlet diameters.

3.4.1.5 Propellant Coupling Mechanism

The propellant coupling mechanism is based on a design by the CALMEC Manufacturing Company for a cryogenic propellant probe - drogue transfer line coupling (Schematically shown in Figure 3-8.) The couplings are of the extending retracting type with the probe on the tanker vehicle retracted until completion of the docking operation. It is then extended by either electrical or hydraulic actuators. Total power required for actuation is between 0.5 and 1.0 kilowatts. Electrical powered actuators appear more feasible for power levels of the above magnitude. If electrical power is used, a locking device is needed to keep the couplings joined. This may be accomplished by use of a worm-and-gear set or by providing a differential area within the pipe such that the propellant pressure over this area provides the required force to maintain the connection.

TANKER VEHICLE PRESSURIZATION SYSTEM SCHEMATIC

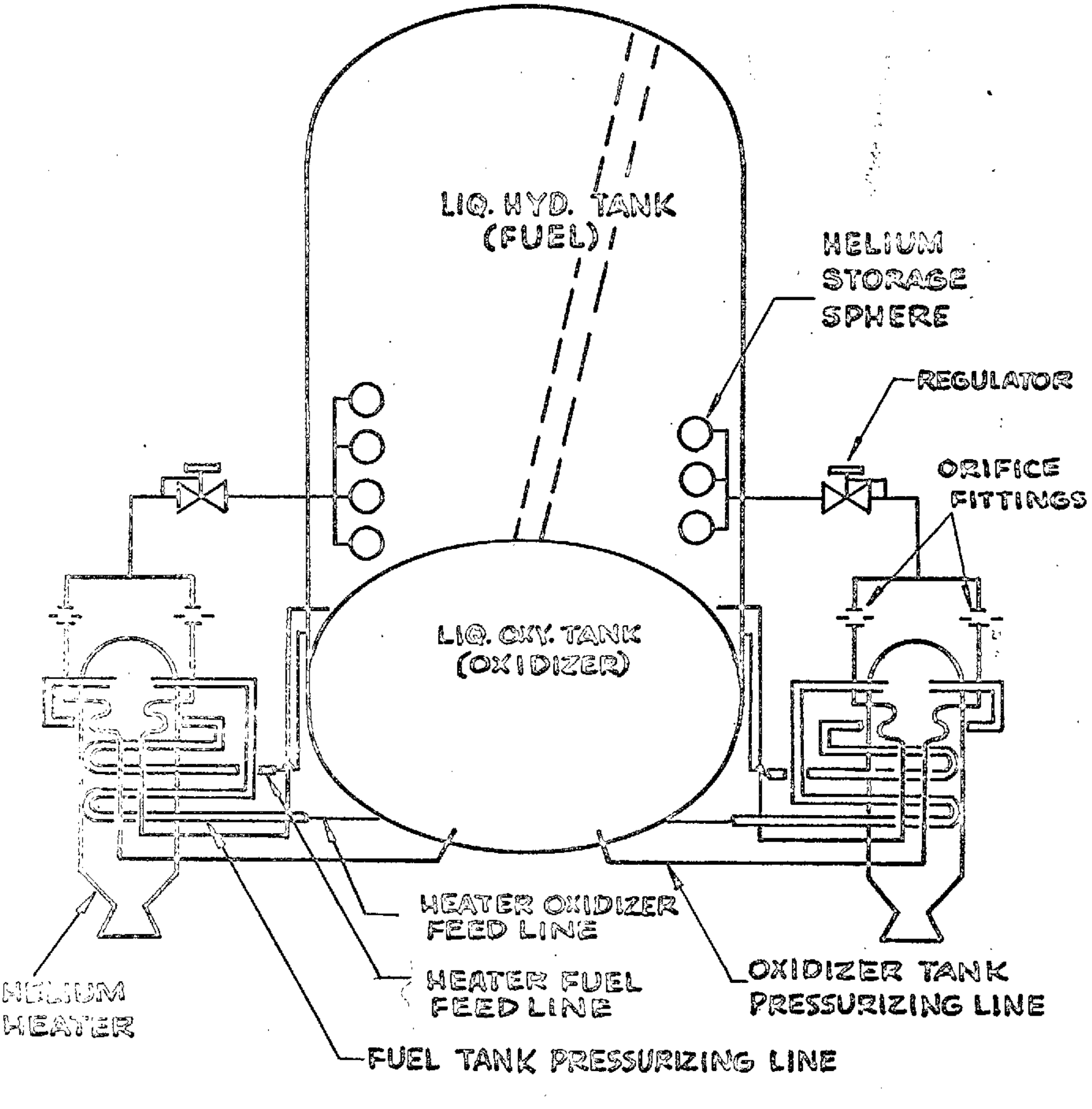


FIGURE 3-5

PRESSURIZATION SYSTEM SCHEMATIC

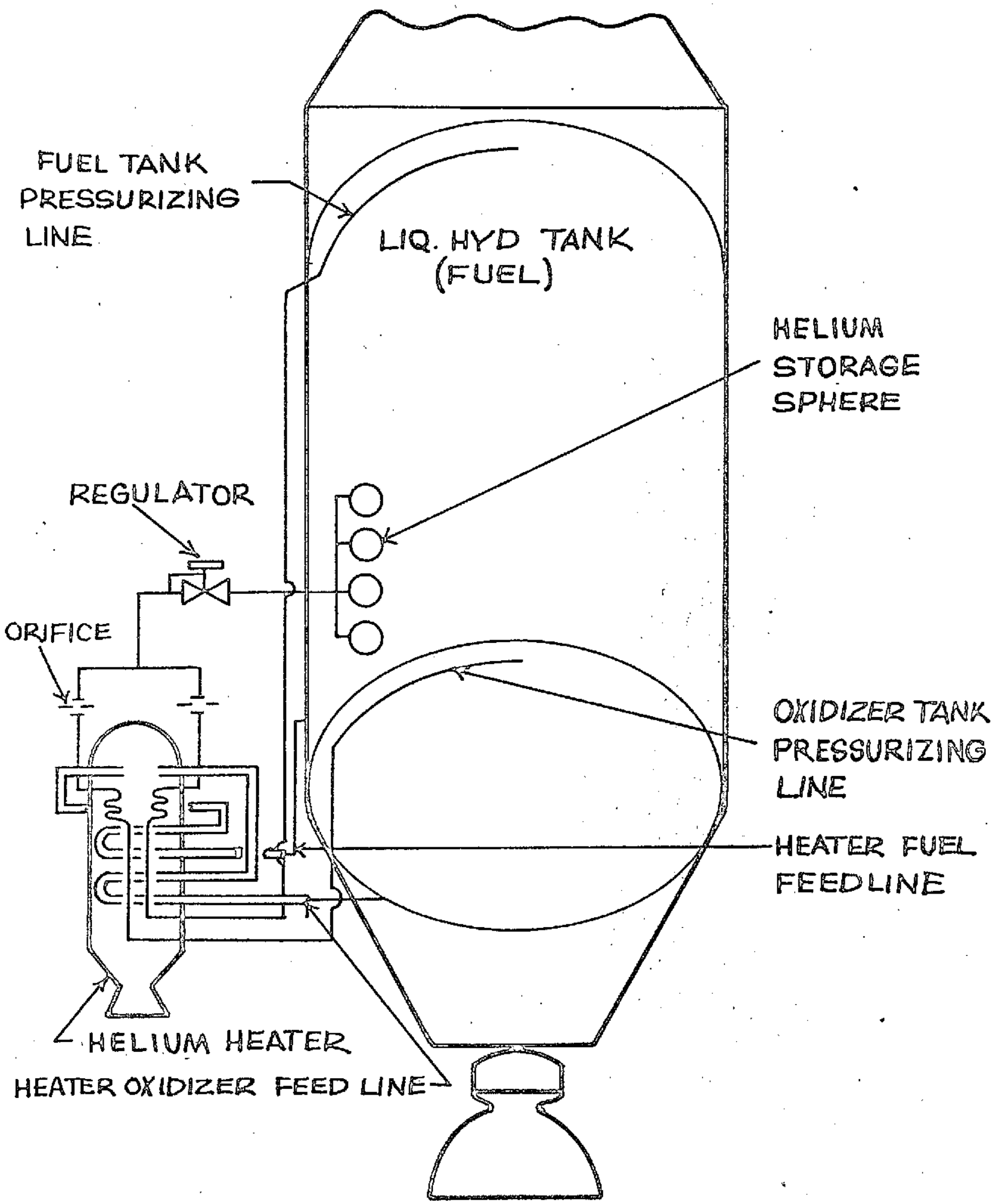
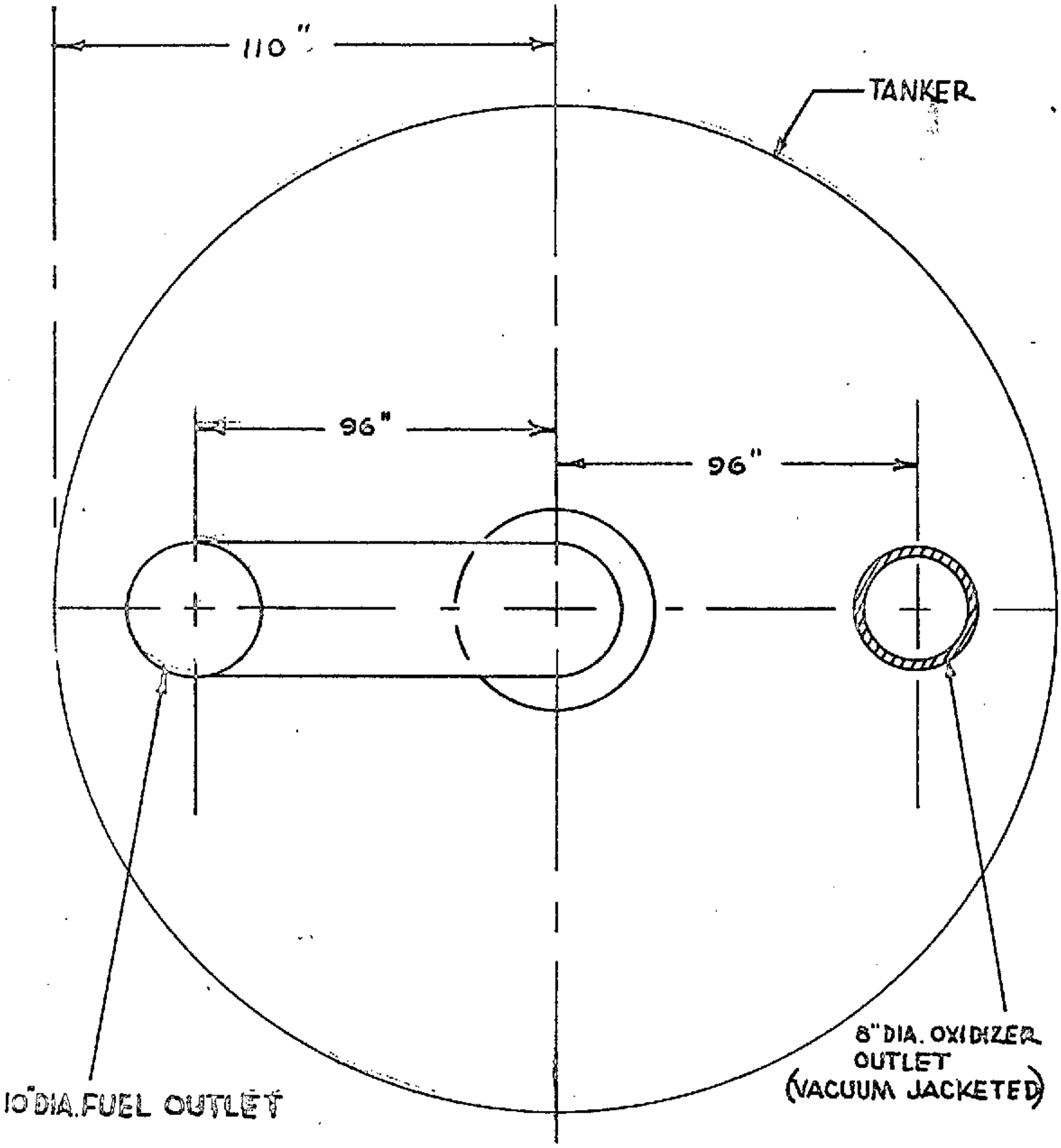


FIGURE 3-6



NOTE: ALL DOCKING DUCT ASSEMBLIES ARE IN THE PLANE OF OUTLETS SHOWN

FIGURE 3-7

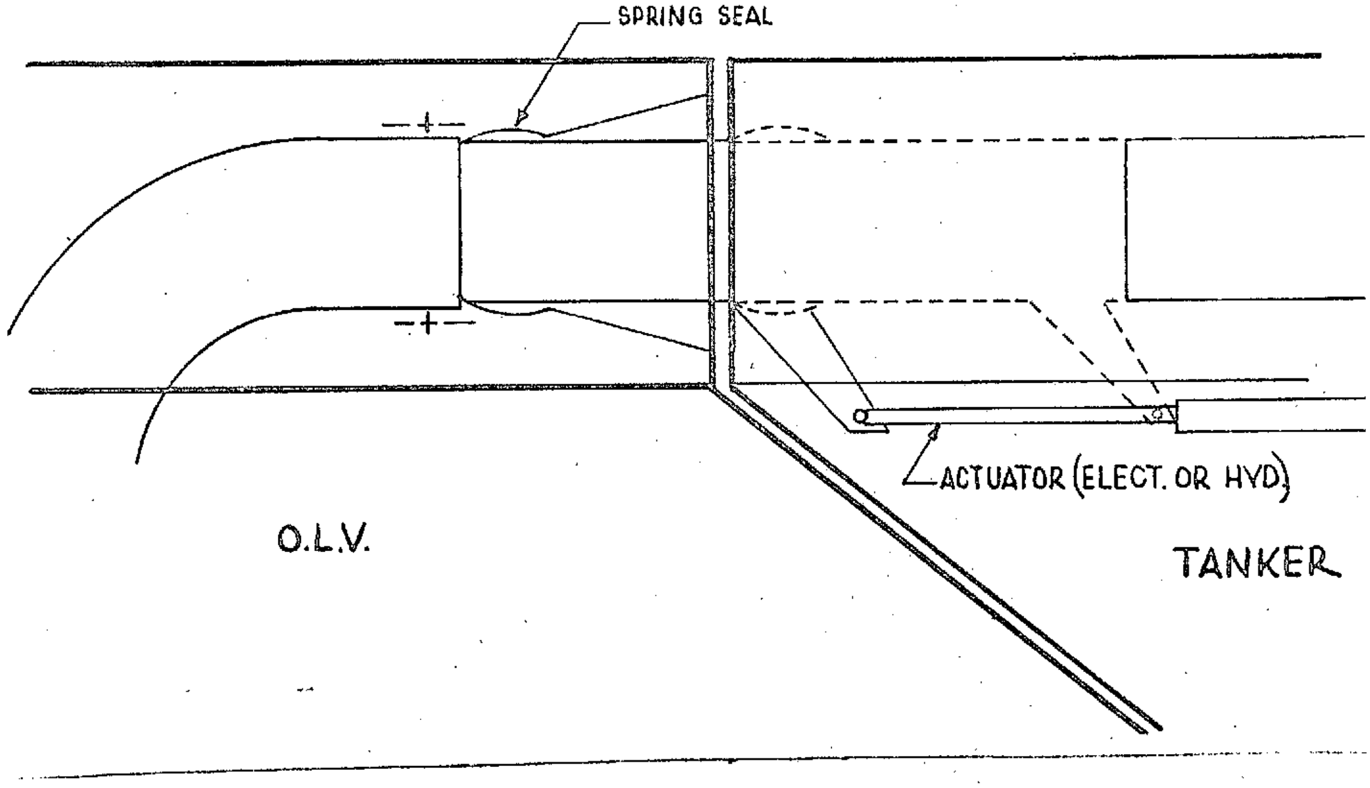


FIGURE 3-8

When the probe has been fully extended into the drogue, the spring hand expands into the sealing ring to seal the connection. Preliminary design study anticipates only minor leakage through this connector.

3.4.2 Operational Procedure

The operational procedure for orbital propellant transfer presented in this study assumes that the docking maneuvers have been completed, the transfer ducts between the orbital launch vehicle and the tanker have been connected, and the over-all system pretransfer checkout procedures have been completed.

3.4.2.1 Sequence of Transfer Operations

To effect propellant transfer the steps shown in Table 3-2 are executed.

3.4.2.2 Vehicle Dynamics During Transfer

A supporting study has been performed to illuminate some of the guidance and control problems associated with an orbital refueling operation. Two major areas were considered, (a) attitude control during the operation, and (b) trajectory perturbations in the linear acceleration mode.

An acceleration must be applied to the vehicles during the fueling operation in order to provide the phase separation required for pumping. A low level thrust is applied in order to provide the required 0.001 g acceleration. The effect of the pumping operation on the vehicle attitude was investigated. The effect of transfer operation on the vehicle pitch moment of inertia and center of gravity location is shown in Figure 3-9 thru 3-12 for the various configurations.

Since no angular momentum is added to the system the momentum of the fuel being pumped must be offset by momentum in the total vehicle. To compute the fuel momentum the transfer plumbing geometry was approximated as in Figure 3-13.

The fluid in the tanks is assumed to be at rest. The angular momentum (H) of the fluid in the pipe about the cg can be shown to be equal to the mass rate ($\dot{m}$), times the product of the pipe length (x) and the distance (r) from the centerline of tanker to centerline of pipe.

$$H = xr\,\dot{m}$$

It is now possible to determine angular displacement induced in the total vehicle for the several configurations proposed. For all configurations $\dot{m} = 6.9$ slugs/sec.

Table 3-2. Sequence of Transfer Operations

ACTION	DIRECTED BY	ACTION BY	MONITORED BY	TIME (SEC)
1. Initiate Propellant transfer operations	Pilot	Pilot	Pilot	T-160
2. Activate the attitude control system for vehicle stabilization	Automatic	Automatic	Pilot Display & Computer	T-155
3. Orient the tanker-orbital launch vehicle system so that its longitudinal axis is in the direction of the desired orbital velocity vector.	Automatic	Automatic	Computer	T-150
4. De-energize and lock closed the liquid oxygen tank and liquid hydrogen tank vent valves (standard on both the tanker vehicle and the orbital launch vehicle booster) and reset tanker LH_2 pressure to 26 psia.	Automatic	Automatic	Pilot Display & Computer	T-90
5. Energize to open at vent pressure both propellant transfer vent valves (Ref. Fig. 23) on the orbital launch vehicle.	Automatic	Automatic	Pilot Display & Computer	T-30
6. Open both propellant control valves in the transfer lines of the tanker vehicle	Automatic	Automatic	Pilot Display & Computer	T-25
7. Start the auxiliary propulsion engines on the tanker vehicle to produce linear acceleration for liquid-vapor phase separation	Automatic	Automatic	Pilot Display & Computer	T-20
8. Ignite the pressurization system helium heater(s) on the tanker vehicle	Automatic	Automatic	Pilot Display & Computer	T-20
9. Open both propellant control valves in the transfer line of the orbital launch vehicle booster. (Initiates propellant transfer)	Automatic	Automatic	Pilot Display & Computer	T-0
10. Open both propellant tank pressurizing valves on the tanker vehicle	Automatic	Automatic	Pilot display & Computer	T-0
11. Monitor the propellant quantity gauges during transfer	Automatic	Automatic	Pilot Display & Computer	T-0
12. Close the pressurizing valves and shut down the helium heater(s) on the tanker vehicle after the propellant transfer has been completed.	Automatic	Automatic	Pilot Display & Computer	T+600
13. Close the propellant control valves in the transfer lines of both the tanker vehicle and the orbital launch vehicle	Automatic	Automatic	Pilot Display & Computer	T+600
14. Shut down auxiliary propulsion engines	Automatic	Automatic	Pilot Display & Computer	T+615
15. Energize to open at vent pressure the standard vent valves and close the transfer vent valves on the orbital launch vehicle.	Automatic	Automatic	Pilot Display & Computer	T+630
16. Open the standard vent valves on the tanker vehicle	Automatic	Automatic	Pilot Display & Computer	T+630
17. Post transfer checkout	Pilot & Computer	Automatic	Pilot Display	T+635
18. Disconnect and restart the auxiliary propulsion system to separate the tanker vehicle from the orbital launch vehicle.	Pilot	Automatic	Pilot Display & Computer	T+935
19. Initiate post separation checkout	Pilot	Pilot	Pilot	T+940

VEHICLE MOI & C.G. DURING PROPELLANT TRANSFER IN ORBIT (S-II WITH S-IVB)

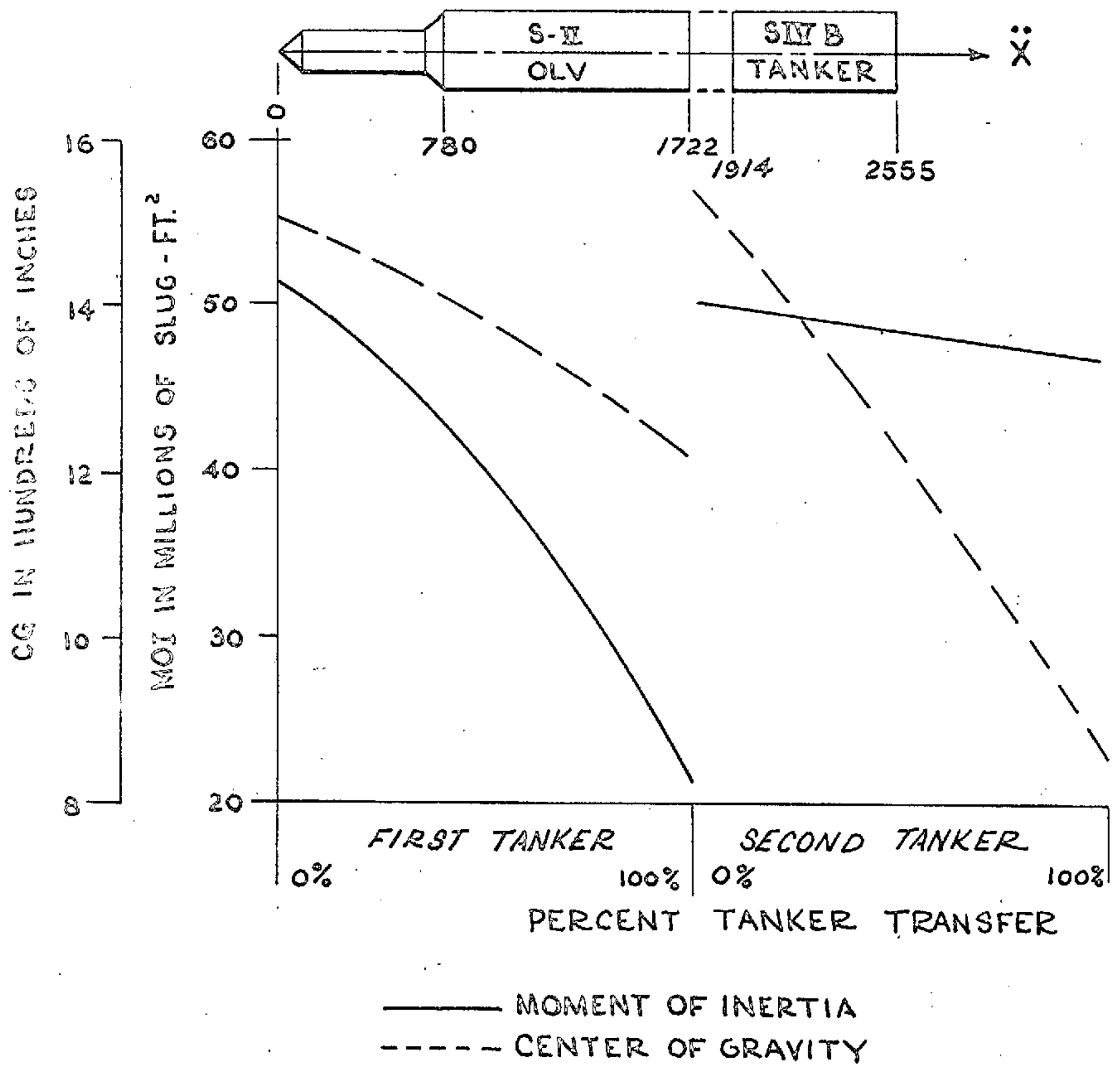


FIGURE 3-9

VEHICLE MOI & C.G. DURING PROPELLANT TRANSFER IN ORBIT (S-IVB WITH S-IVB)

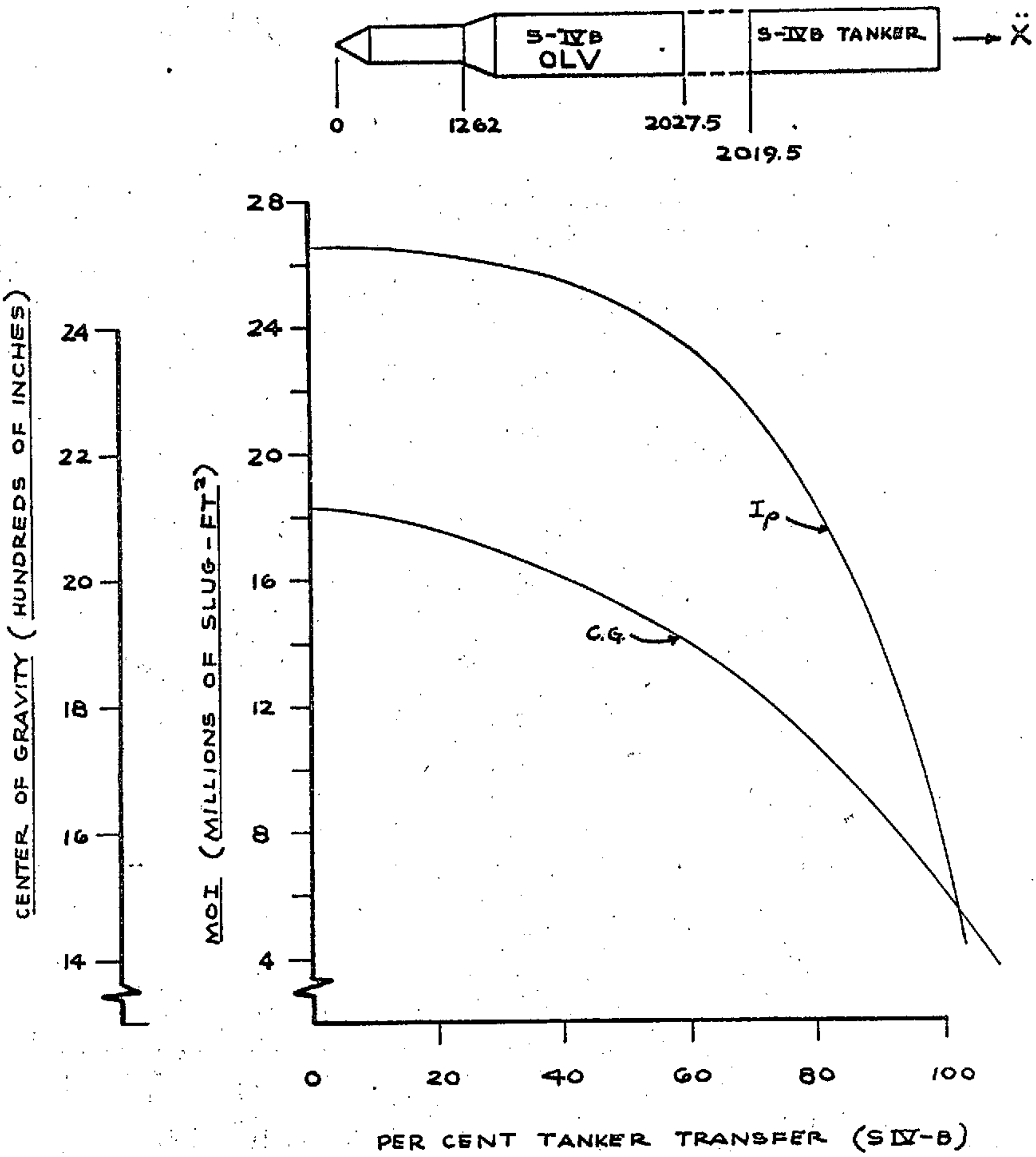


FIGURE 3-10

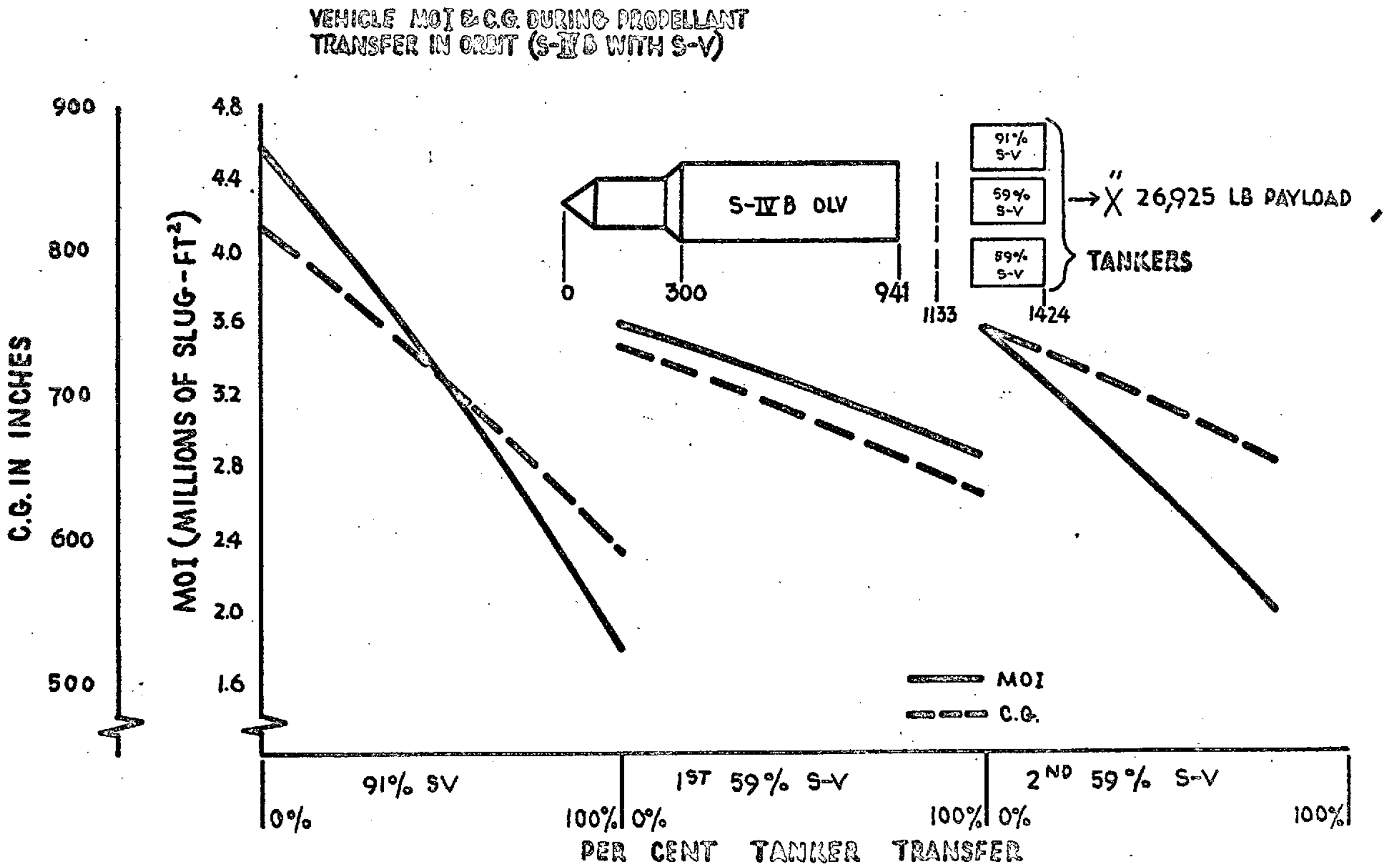
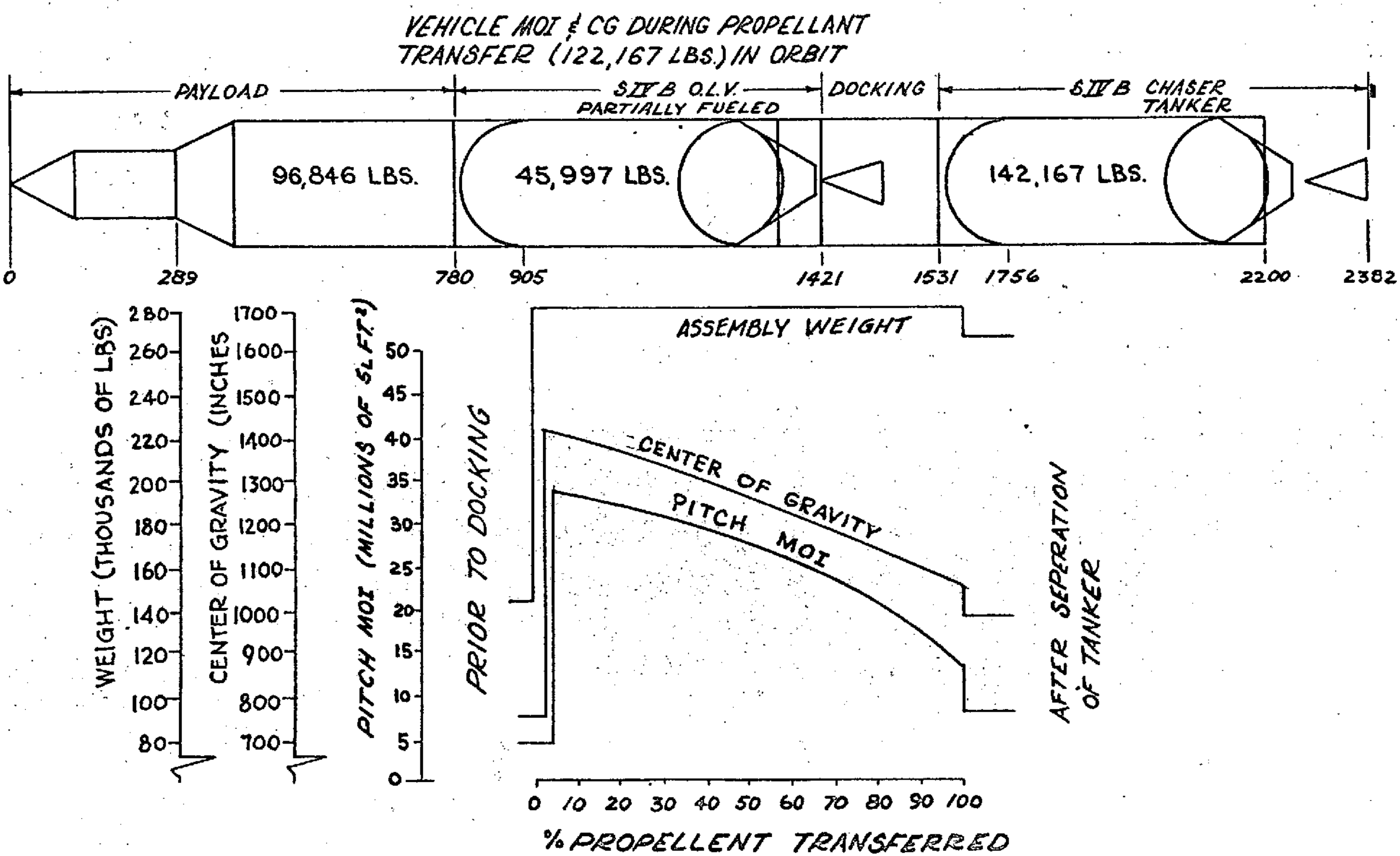

VEHICLE MOI & C.G. DURING PROPELLANT TRANSFER IN ORBIT (S-IVB WITH S-V)
C.G. IN INCHES
900
800
700
600
500
MOI (MILLIONS OF SLUG-FT²)
4.8
4.4
4.0
3.6
3.2
2.8
2.4
2.0
1.6
S-IVB DLV
0
300
941
1133
1424
91% S-V
59% S-V
59% S-V
26,925 LB PAYLOAD
TANKERS
MOI
C.G.
91% SV
1ST 59% S-V
2ND 59% S-V
0%
100%
0%
100%
0%
100%
PER CENT TANKER TRANSFER

FIGURE 3-11



VEHICLE MOI & CG DURING PROPELLANT TRANSFER (122,167 LBS.) IN ORBIT
PAYLOAD
SIVB O.L.V. PARTIALLY FUELED
DOCKING
SIVB CHASER TANKER
96,846 LBS.
45,997 LBS.
142,167 LBS.
0
289
780
905
1421
1531
1756
2200
2382
WEIGHT (THOUSANDS OF LBS)
CENTER OF GRAVITY (INCHES)
PITCH MOI (MILLIONS OF SL FT²)
PRIOR TO DOCKING
ASSEMBLY WEIGHT
CENTER OF GRAVITY
PITCH MOI
AFTER SEPERATION OF TANKER
% PROPELLENT TRANSFERRED

FIGURE 3-12

SCHEMATIC OF TRANSFER GEOMETRY FOR VEHICLE DYNAMIC EVALUATION

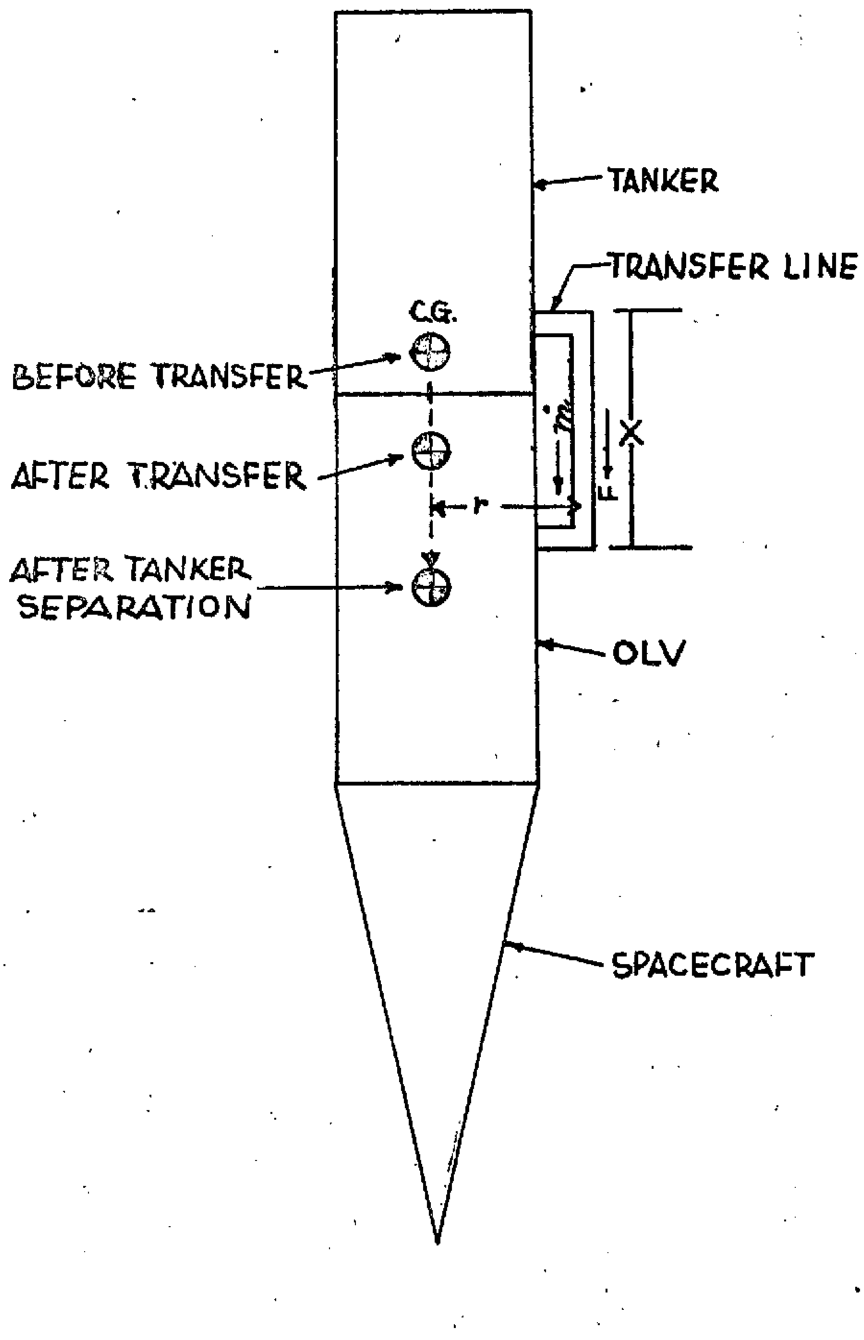


FIGURE 3-13

(a) Typical OLO Program - Lunar Landing Mission

$H = 3,450$ slug ft.^{2}/sec.

Transfer Time = 790 sec.; in two segments

$\omega_{max} = .17$ mr/sec. where ω_{max} is the maximum turning rate during fueling in milliradians per sec. (.001 rad/sec)

$\theta_t = 81$ mr $= 4.7°$

θ_t = total vehicle rotation during fueling

(b) Accelerated OLO Program - Lunar Landing Mission

$H = 4,000$ slug ft.^{2}/sec.

Transfer Time = 365 sec.

$\omega_{max} = .35$ mr/sec.

$\theta_t = 94$ mr $= 5.3°$

(c) Accelerated OLO Program - Lunar Orbit Mission

Refueling is in three stages

$H = 1,600$ slug ft.^{2}/sec.

Transfer Times = 80 sec., 52 sec., and 52 sec.

$\omega_{max} = .95$ mr/sec.

$\theta_t = 108$ mr $= 6.5°$

From the above angular displacements it is seen that the disturbing effects upon the vehicle induced by propellant transfer are not of a serious nature. It is felt that they will impose no additional requirements upon the attitude control system.

Little is known about the dynamics of fuel sloshing under such low acceleration conditions. However, it is felt that the very low sloshing frequency, (on the order of 10^{-2} cps), will probably prevent any serious oscillation.

3.4.2.3 Orbit Change During Transfer

The use of linear acceleration for phase separation during pumping will, of course, perturb the original orbit of the vehicle. Assuming 0.001 g acceleration for 10 min., (18 ft/sec. velocity increment), and a 300 nautical miles circular orbit these perturbations may be computed. They are found to be as follows:

(a) Change in perigee altitude, Δr_p

$$\frac{\delta r_p}{\delta V} = 0.3 \text{ n.mi./ft./sec.}$$

$$\Delta r_p = 5.4 \text{ n.mi.}$$

(b) Change in eccentricity, Δe,

$$\frac{\delta e}{\delta V} = \frac{2}{r_p} \frac{\delta r_p}{\delta V} = .162 \times 10^{-4}/\text{ft/sec}$$

$$\Delta e = .0003$$

(c) The rotation of the line of apsides will be 24°. However, the flight path angle at the original argument of perigee will be only 0.123 mr because of the very small eccentricity.

These results indicate that the orbital perturbations will be small in terms of their effect on subsequent system performance. The guidance system must operate during the fueling operation, of course, to maintain system accuracy. The guidance system aboard the spacecraft will be utilized.

3.4.3 Man's Role in Transfer Operations

The propellant transfer system discussed was designed to operate in an automatic mode, primarily. However, the operation is initiated by man after he is assured of the success of the docking operation. Similarly when the propellants have been transferred, the disconnect and separation of the tanker from the space vehicle may be initiated by man.

A pretransfer checkout operation will be performed by man. This will normally be performed in the command module by use of remote sensing devices located in the tanker, the docking interface, and the OLV. The items are:

(a) Verification of successful docking operation with assurance of proper alignment.

(b) Verification of proper pressure levels in the tanker and OLV.

(c) Verification of proper temperature levels in the tanker propellants and the OLV tanks.

(d) Verification of proper propellants loadings aboard the tanker vehicle (±0.5% in a small g field; ±2% in a "0" g field) through a propellant tank electrical capacitance measurement.

(e) Similar verification of the auxiliary propulsion system propellant supply (storable hypergolic).

(f) Pretransfer firing of the auxiliary propulsion engines (2 seconds firing).

In the normal operation, man will also perform the role of monitor of the transfer system and be supplied with manual override controls for the following operations:

(a) Start and shutdown of the auxiliary propulsion system.
(b) Extension and retraction of the propellant coupling probe.
(c) Open and closing of propellant control valves.
(d) Start and shutdown of the helium heater pressurization operation.
(e) Start and shutdown of the attitude control system aboard the tanker (as well as the Space Vehicle).
(f) Operation of the vent valves on both the tanker and Space Vehicle.

It is anticipated that these control operations could be performed by man in the event of failure of the automatic control system.

In addition, man will be capable of certain manual operations exterior to the command module. This will require man to place himself on the vehicle in the region of the docking interface. For this purpose, attach points for the man should be provided on the docking structure outer surface. The operations which man may perform here are:

(a) Inspection and verification of the docking mechanism completed operation.
(b) Inspection and verification of the propellant coupling completed operation extension and retraction.
(c) Manual operation of propellant control valves in the transfer lines between the tanker and space vehicle through an access panel in the docking structure (a T-bar handle tool may be employed).
(d) Manual alignment and extension or retraction of the propellant coupling probe (some type of crowbar lever may be employed).
(e) Closing and opening of locking latches in the docking mechanism and the propellant coupling mechanism.

Man is thus brought into the system primarily as a command and monitoring role but also with the capability of manual control override and actual manual operation of certain parts of the propellant transfer system.

3.4.4 Safety Considerations

The inclusion of man in the propellant transfer system requires an evaluation of the degree of safety associated with the operation. Propellant loading on earth is considered a potentially dangerous operation. In orbital operations this does not appear to be the case if certain precautions are observed in the design and operation of the system. Also, there are several safety factors intrinsically available due to the orbital environment.

(a) Extremely rapid dispersion and expansion of any propellant leakage to concentrations far below pressures supporting combustion.
(b) Absence of a natural reactant agent for supporting combustion in the environment.
(c) Absence of corrosive or electrical conducting vapor or fluid (e.g., atmospheric water) in the environment (minimizing the possibility of electrical shorts).

(d) Absence of an ice-forming (on cryogenic equipment) vapor, or fluid, in the environment.

(e) Absence of an environmental media for conducting shock waves (as from an explosion of the tanker vehicle, etc.) if an explosion should occur. This means that "overpressure" structures such as blockhouses are not needed for protection from an explosion.

(f) Other safety factors not directly related to propellant transfer operations itself, but to orbit launch operations, i.e., no catastrophic failure in the event of thrust termination at "lift off," no range safety problem (destruct system not required), no wind conditions imposing control problems, etc.

In conjunction with the safety factors arising from the orbital environment, certain design features of the recommended transfer system (with end-to-end docking) are:

(a) Location of the manned command module, which is also the abort vehicle, at the opposite end of the vehicle's propellant transfer area. This provides distance and interposing structure between the manned module and any hazardous mishap in the transfer area. It also allows a direct escape route in the event of catostrophic failure.

(b) Location of the LO_2 and LH_2 transfer coupling lines at opposite sides of the vehicle and near the outside of the vehicle so that any leakage may be rapidly dispersed through the access openings or open structure associated with the docking and transfer area.

(c) The end-to-end docking arrangement minimizes the possibility of propellant tank collision and rupture during the docking and departing operation.

(d) The propellant transfer line couplings and operations do not begin until a successful rigid docking arrangement has been secured. This minimizes the possibility of the transfer lines parting during the transfer operation, releasing large masses of propellants which might give rise to dangerous concentrations temporarily.

(e) Isolation of the auxiliary propulsion system from the remaining components such as the propellant couplings, etc. This assures complete dissipation of the exhaust gases before they reach the area of transfer operations.

Additional considerations are the pretransfer checkouts performed before the transfer operation is initiated. This assures that the major subsystems are operating. In addition the manual control override allows the man to halt the automatic sequence of operations at discrete points in the event of failure, and to shut off the propellant flow, pressurization system or auxiliary propulsion system before a dangerous condition may evolve. In cases where this may exceed the man's capability to detect and respond to a catastrophic failure, an automatic abort system should be provided to either: (a) separate and remove the space vehicle (including the OLV) from the tanker, or (2) separate and remove the manned module plus service module and re-entry section from the space vehicle. The determination of which of the two courses should be selected will depend upon the nature of the failure and its resultant effects, and whether the OLV may be salvaged in a useable condition and a spare tanker is provided. Manual abort should also be provided for cases not involving catastrophic failure.

The capability of manual operations in the docking area should be regarded as an emergency procedure and necessarily entails risk and hazards for the man performing this task.

3.4.5 System Design and Analysis

The initial problem encountered in the vehicle subsystem design is the behavior of liquid propellants under orbital conditions of zero "g" gravity.

Theoretical studies and small scale tests indicate that the ullage bubble collects at the center of the tank. This imposès a major vent and transfer problem. In order to achieve efficient propellant transfer, a phase separation should be provided. When this condition has been established, the transfer operation can be initiated.

One of the most convenient ways to separate liquid from vapor is by means of linear acceleration. A low "g" level (10^{-3} g) was selected to fulfill the first condition for two reasons, acceptable settling time of the phase separation and the total weight of the auxiliary hypergolic propulsion system which provides the linear acceleration. The propellant transfer will be performed by a propellant tank pressurizing system. Thus the proposed method is a combination of two vehicle subsystems.

The analysis in this study substantiates the feasibility of this system. A fundamental and practical analytical approach was taken to avoid undue complexity and permit a wider scope within the study duration. However, full consideration was given to the operational reliability requirements.

The problem as previously stated is to transfer cryogenic propellants (LH_2 and LO_2) from the tanker into the OLV receiving tanks. Due to losses through trapped propellant, boil-off, etc., the tanker loading should exceed the required OLV loading by 5% to account for these losses.

This requirement applies to simultaneous propellant mass transfer operation for the following total propellant weights and design criteria which were stated in the work statement:

(a) 200,000 lbs (total useable propellants)
100,000 lbs
20,000 lbs
(5:1 mixture ratio)

(b) Maximum pressure limit is dictated by the tankage structural limitation.

(c) Minimum pressure limit imposed by the saturation pressure (below which excessive evaporization occurs)

These design criteria automatically determine the minimum transfer time and the total auxiliary propulsion required.

The tabulation (Figure 3-14) shows the relationship between the design variables and the characteristic data of the selected system. For

FIGURE 3-14

LINEAR ACCELERATION TRANSFER SYSTEM
WEIGHT DATA
(DOCKING COMPLETED)

General	Acceleration (g_o)	$\frac{1}{1000}$	$\frac{1}{500}$	$\frac{1}{100}$	$\frac{1}{1000}$	$\frac{1}{500}$	$\frac{1}{100}$	$\frac{1}{1000}$	$\frac{1}{500}$	$\frac{1}{100}$
	Transfer Time (sec.)		600			300			51	
	Propellant Transferred (lbs.)		200K			100K			20K	
	Payload (lbs.)		160K			30K			20K	
	OLV (lbs.) - Empty, insulated & partially fueled		25K			18K			18K	
	Tanker (lbs.) - (Includes 30 day insulation and meteoroid protection)		S-IVB Tanker 28,605			S-IV Tanker 18,250			S-V Tanker 2,530	
Pressurization System	Helium Heater (lbs.)		80			80			40	
	Helium, lines, etc.		2,115			1,360			385	
	Total		2,195			1,440			425	
Auxiliary Propulsion System	Thrust Required (lbs.)	417	836	4,440	168	336	1,700	51	122	510
	Selected Motors	2 x 225	2 x 425	2 x 2100	2 x 90	2 x 175	2 x 850	2 x 35	2 x 70	2 x 300
	Bi-Propellants* (lbs.)	950	1,900	8,200	175	355	1,630	15	30	122
	System Weight (lbs.) - (Includes hardware and propellant)	1,200	2,110	8,860	310	505	2,200	45	65	250
Total Assembly Weight (lbs.) (During Transfer)		417,000	417,910	424,660	168,000	168,195	169,890	51,000	51,020	51,205

*Propellant for venting operations, pretransfer test firing, settling, transfer operation, and separation.

analysis the transfer of 200,000 lbs of propellant will be considered.

The following assumptions are established for the mission conditions immediately prior to transfer:

(a) Docking maneuvers are complete.
(b) The tanker is coupled to OLV with a rigid attachment and the propellant transfer lines are connected.
(c) In orbit, the propellant tank pressure will be controlled by venting to the following conditions:

(1) Orbital Tanker (up to 30 days in a parking orbit)

LH_2 Tank	LO_2 Tank
P_h = 20 psia (saturated)	P_o = 16 psia (saturated)
T_h = 39°R (saturated)	T_o = 166°R (saturated)

(2) Orbital launch vehicle (1 day parking orbit prior to propellant loading)

LH_2 Tank	LO_2 Tank
P_h = 24 psia (saturated)	P_o = 30 psia (saturated)
T_h = 40°R (saturated)	T_o = 175°R (saturated)

Immediately prior to propellant transfer, the following adjustments will be made to the tanker:

(a) Increase LH_2 tank 6 psi to 26 psia
(b) Increase LO_2 tank 20 psi to 36 psia

The orbital launch vehicle vents will remain at 24 psia for the fuel tank and 30 psia for the oxygen tank.

These tank pressures will create the necessary transfer pressure differentials of 2 psi and 6 psi for the fuel and oxidizer, respectively, and will preclude the possibility of propellant boiling in the orbital launch vehicle receiver tank.

After transfer the following conditions will exist in the orbital launch vehicle tanks:

(a) Fuel Tank

P_h = 24 psia (saturation pressure - 20 psia)

(b) Oxygen Tank

P_o = 30 psia (saturation pressure - 16 psia)

To meet the engine starting requirements, a pressure increase on both propellant tanks of 3 psi will be necessary on the orbital launch vehicle. The final tank pressure 27 psi for LH_2 and 33 psi for LO_2, will be sufficient to overcome propellant feed line friction and acceleration pressure loss and to provide adequate net positive suction head pressures at the turbopump inlets.

3.4.5.1 Propellant Transfer Lines

The orbital launch vehicle booster propellant transfer lines (inlet) were established as ten inches and eight inches in diameter for liquid hydrogen and liquid oxygen, respectively. To minimize line losses, the tanker vehicle transfer line (outlet) diameters were made the same as those of the orbital launch vehicle booster. Additional design information of the propellant transfer ducts are summarized below.

	Fuel Line	Oxidizer Line
Length (Ft.)	31	60
Flow rate (Lb/Sec.)	56.8	276
Velocity (Ft/Sec.)	24.5	11.5
Pressure drop (psi)	2.7	6.5

Orbital launch booster tank pressures of 24 psia and 30 psia for liquid hydrogen and liquid oxygen, respectively, were assumed to be sufficient to prevent propellant vaporization during transfer. Considering the pressure drop in the lines, the minimum tanker vehicle tank pressures required for propellant transfer are 26 psia and 36 psia for the fuel and the oxidizer, respectively.

3.4.5.2 Pressurization System

Heated helium gas was selected as the pressurant for pressurizing both propellant tanks. A pressurant gas compatible with both propellants was desirable to simplify tanker systems and loading logistics. Several types of pressurant gases are available for pressurizing the propellant tanks; however, the utilization of either oxygen or nitrogen for oxidizer tank pressurization would result in a considerable weight penalty.

Figures 3-15 through 3-20 show the amount of pressurant gas required to transfer 200K, 100K and 20K of propellants using (1) cold helium and helium heater system and (2) ambient helium. As shown by these graphs, the cold helium and helium heater pressurization system weighs considerably less than the ambient helium system. The temperature to which the helium is heated was limited to 200° R so as to minimize heat transfer to the propellants during transfer and ullage pressure decay. A higher gas temperature would result in a lower specific pressurization system weight; however, higher tank pressures would perhaps be required in the orbital launch vehicle booster to prevent propellant vaporization during transfer. Conversely, higher tank pressures (and structure weight) would be required in the tanker vehicle to transfer propellants.

Propellant transfer system weight and required tanker propellant tank ullage pressures as a function of simultaneous transfer time is shown in Figures 3-21 and 3-22 for 200K and 100K of propellants. The lowest transfer system weight for the 200K propellant system occurs at a transfer time of approximately eight minutes. As noted, this transfer time requires

COLD HELIUM-HEATER SYSTEM WEIGHT TO TRANSFER 200K PROPELLANTS

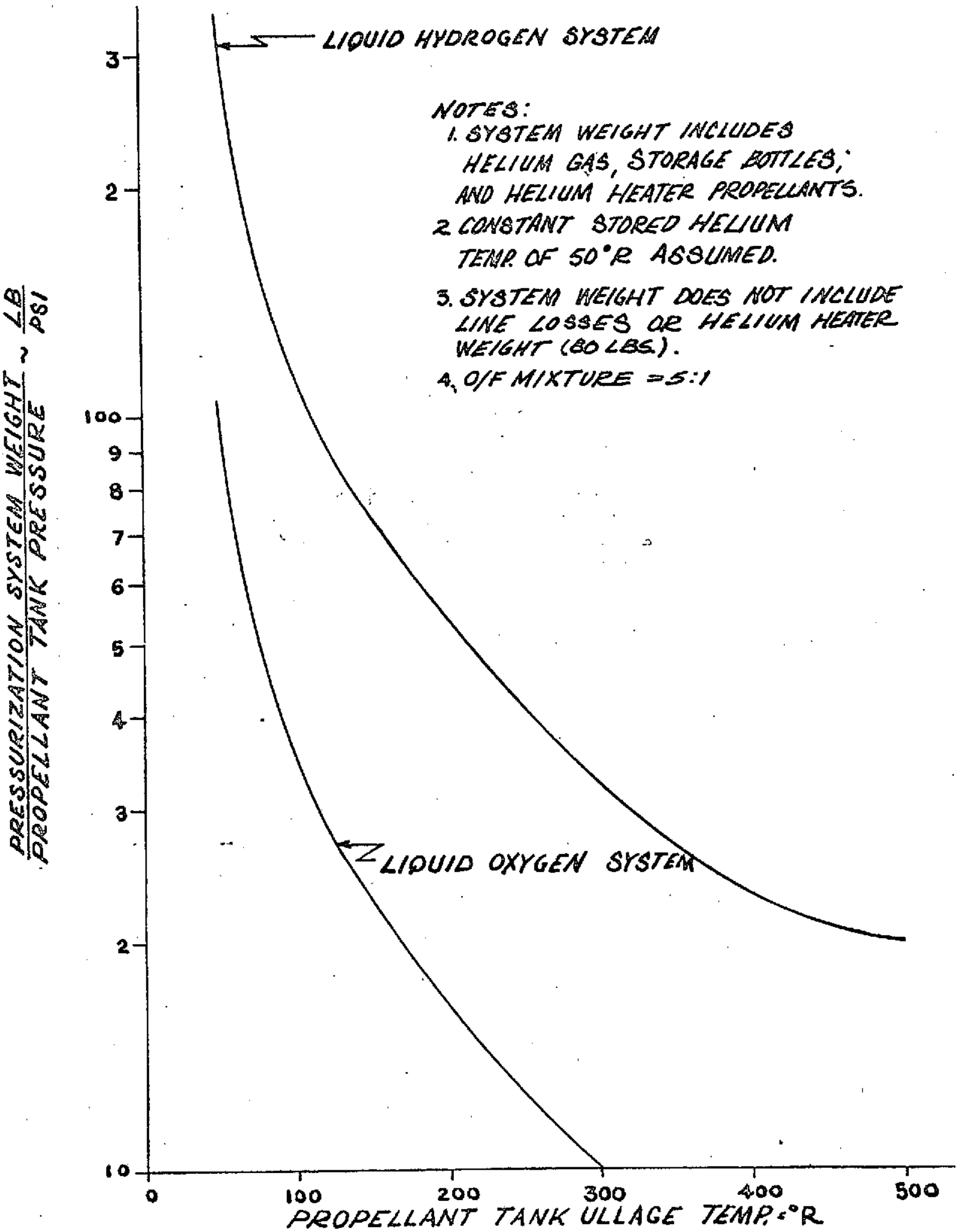


FIGURE 3-15

AMBIENT HELIUM SYSTEM WEIGHT
TO TRANSFER 200 K PROPELLANTS

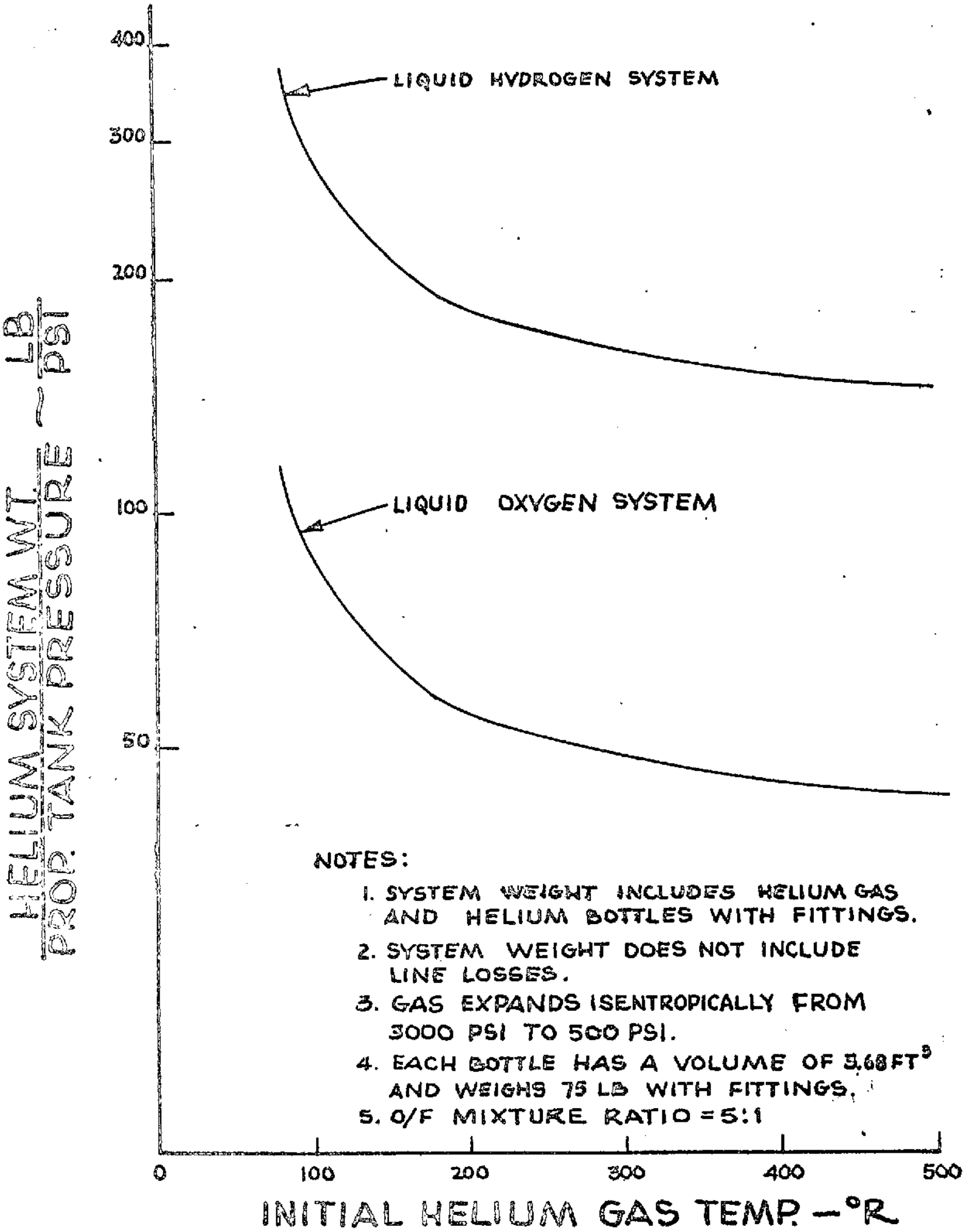


FIGURE 3-16

COLD HELIUM-HEATER SYSTEM WEIGHT TO TRANSFER 100 K PROPELLANTS

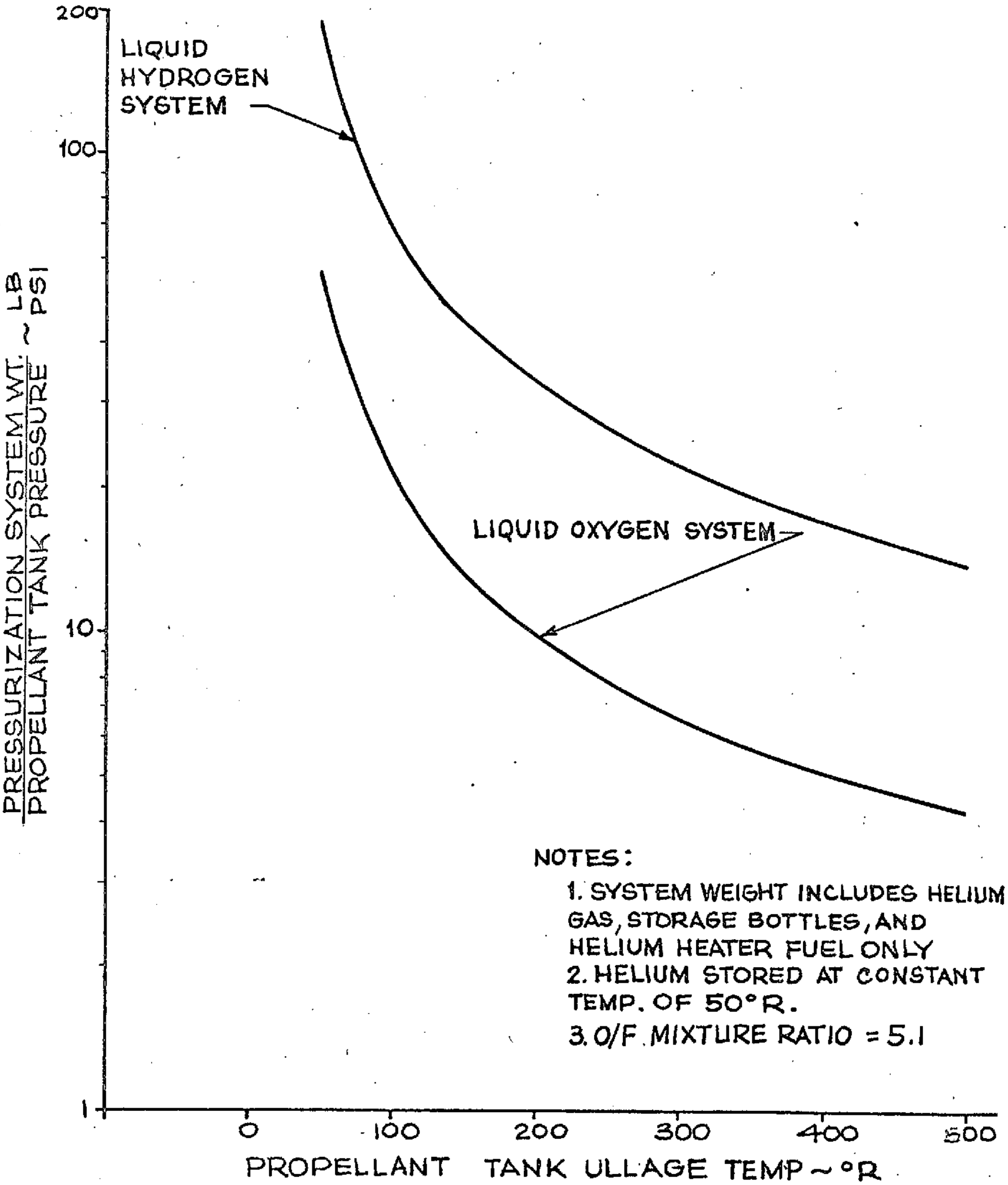


FIGURE 3-17

AMBIENT HELIUM SYSTEM WEIGHT TO TRANSFER 100 K PROPELLANTS

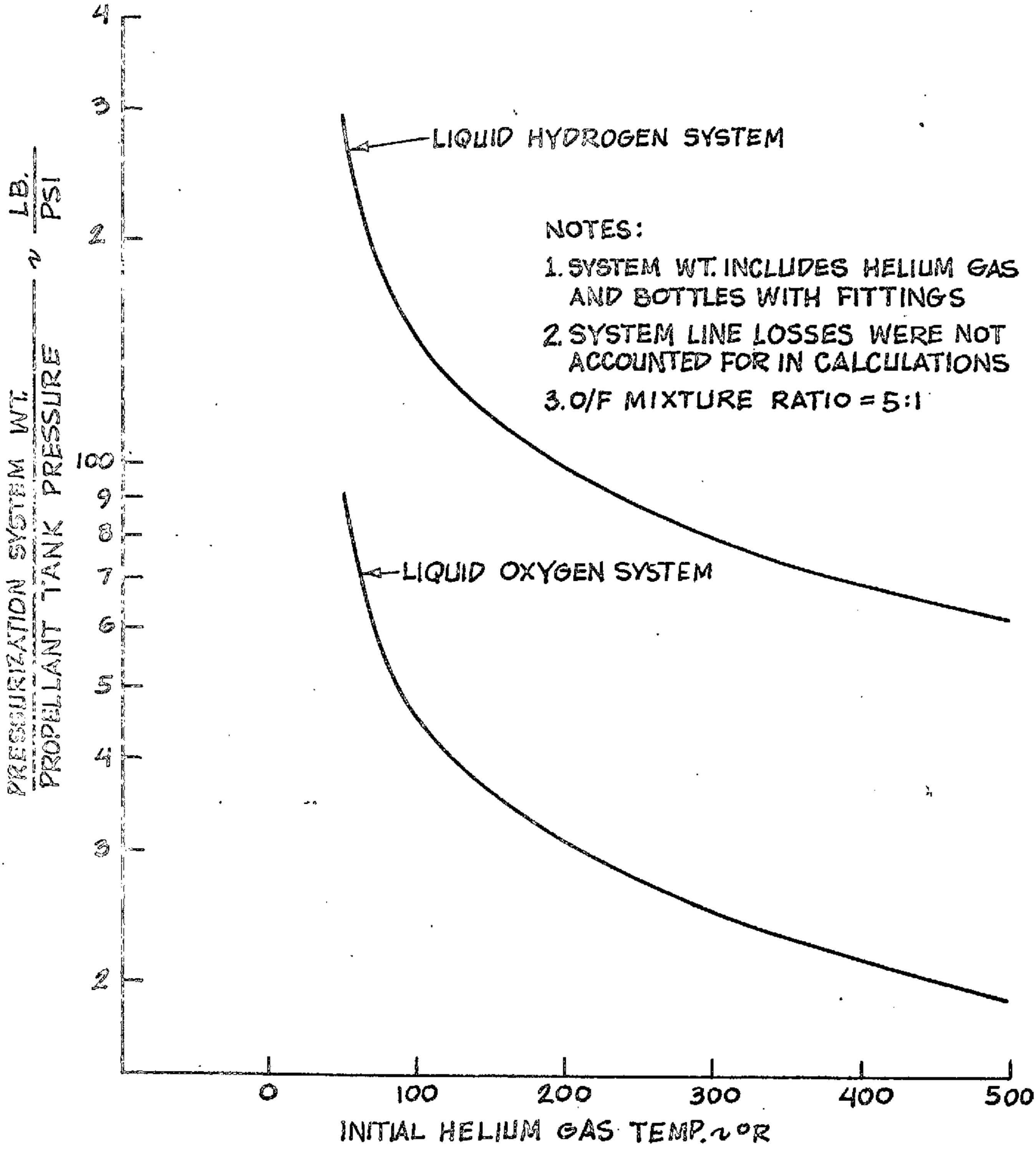


FIGURE 3-18

COLD HELIUM-HEATER SYSTEM WEIGHT TO TRANSFER 20 K PROPELLANTS

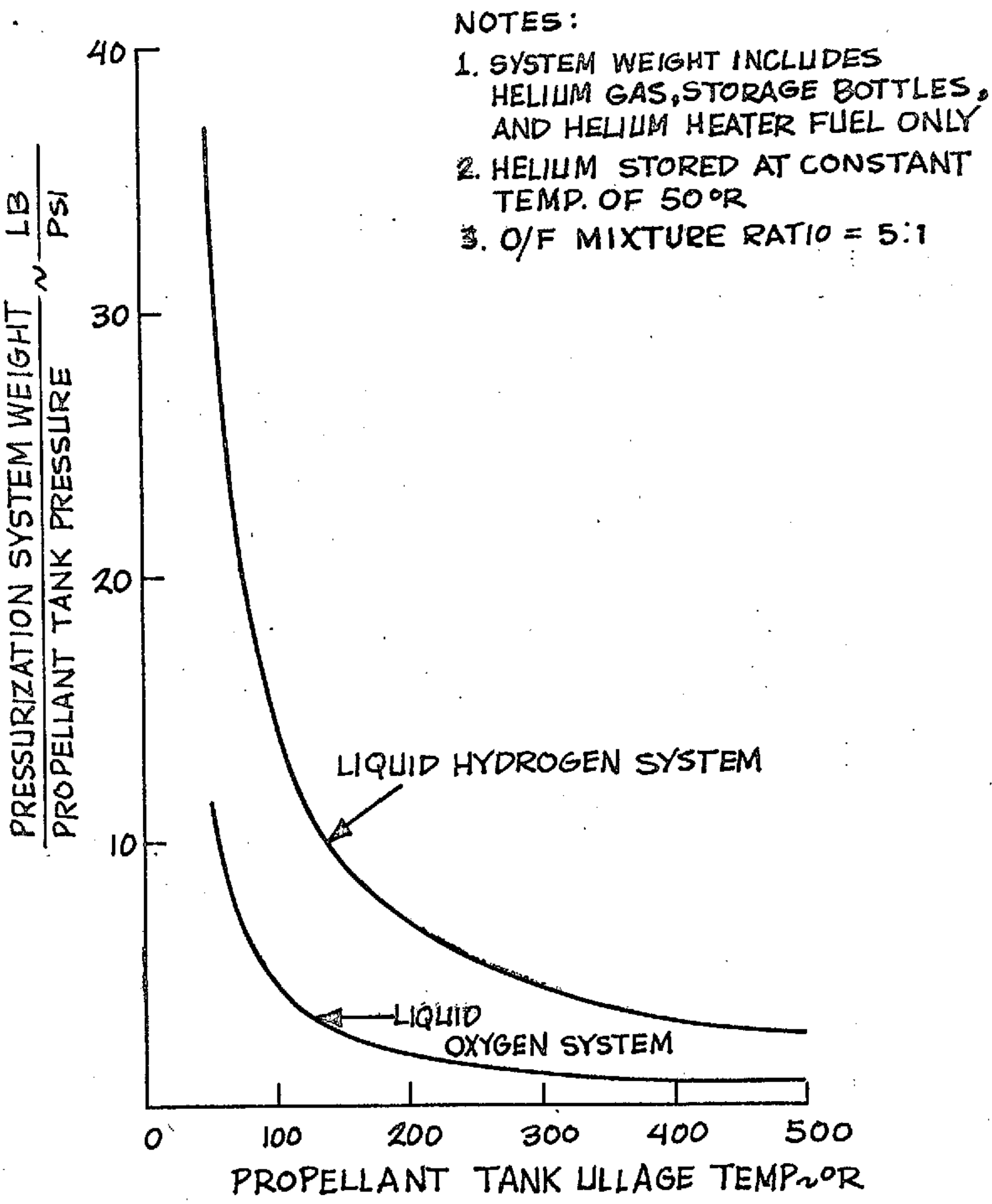


FIGURE 3-19

AMBIENT HELIUM SYSTEM WEIGHT TO TRANSFER 20 K PROPELLANTS

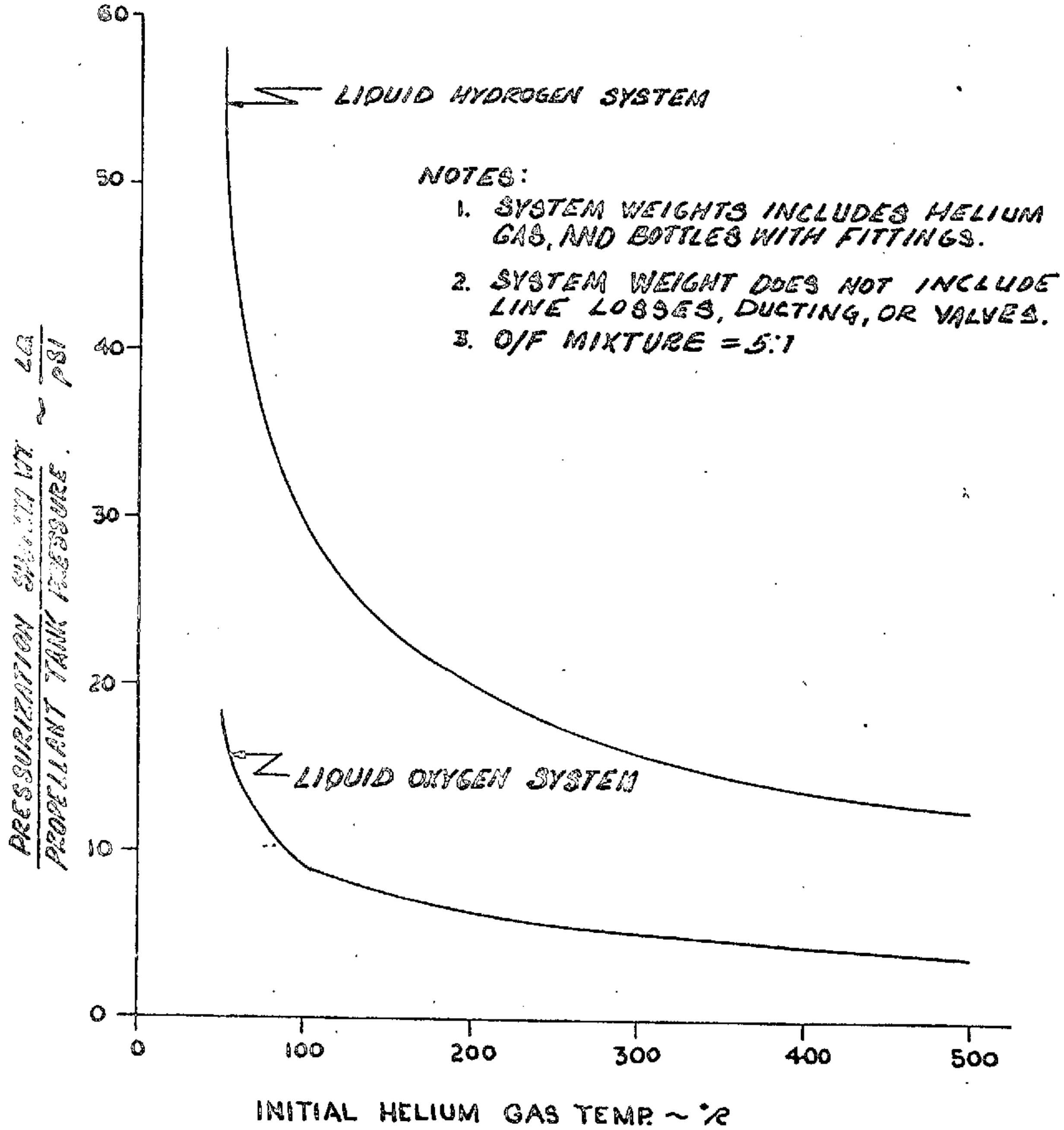


FIGURE 3-20

TRANSFER SYSTEM WEIGHT AND TANK ULLAGE PRESSURE VS. TIME TO TRANSFER 200 K PROP.

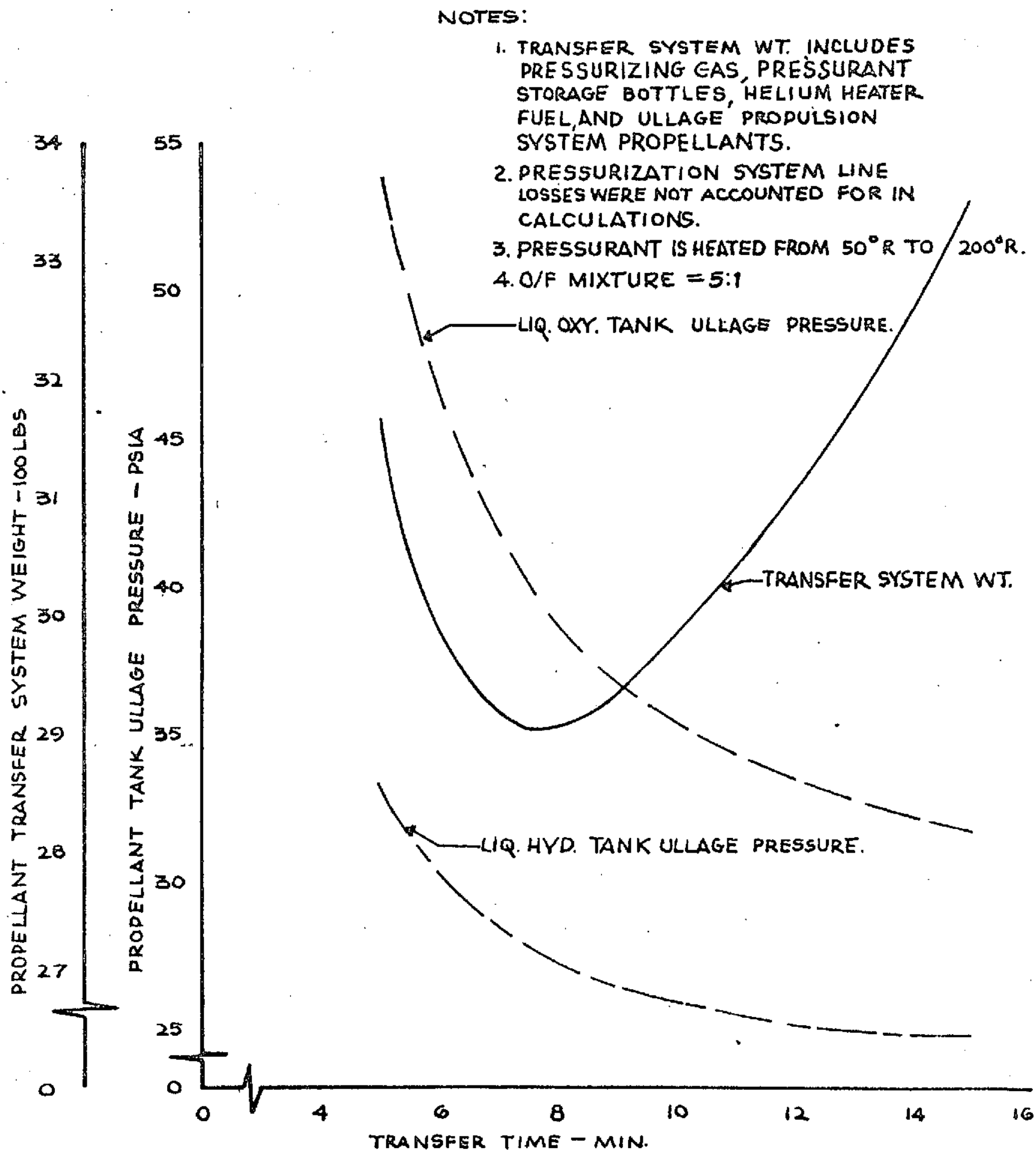


FIGURE 3-21

TRANSFER SYSTEM WEIGHT & TANK ULLAGE PRESSURE Vs TIME TO TRANSFER 100K PROPELLANTS

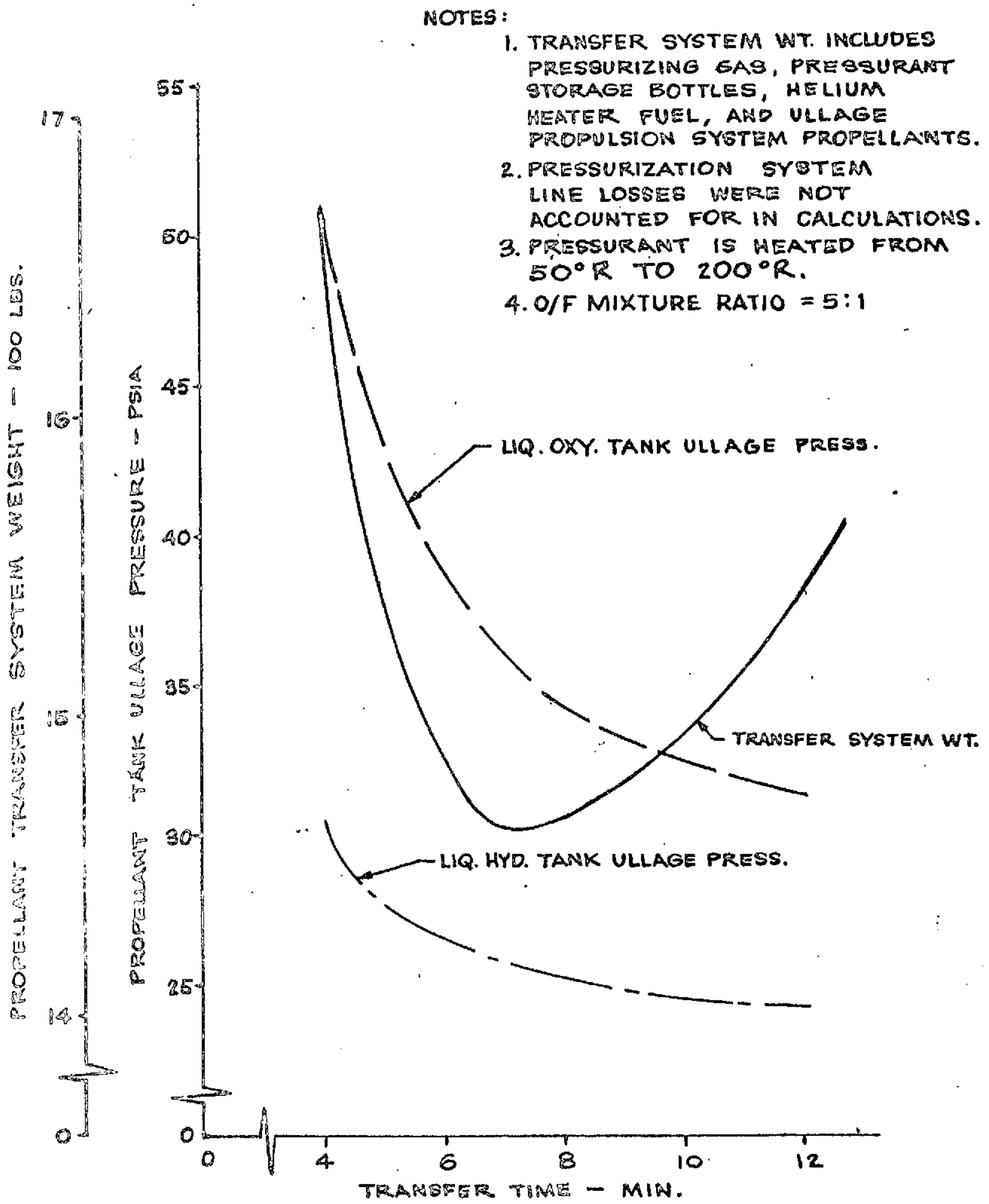


FIGURE 3-22

a liquid oxygen tank ullage pressure of 39 psia; however, the structural design of the tank limits the maximum operational pressure to 36 psia, which results in a transfer time of approximately ten minutes. The corresponding liquid hydrogen tank ullage pressure for simultaneous propellant transfer is 26 psia. For the 100K propellant system, the above tank ullage pressures coincide with the optimum transfer system weight, which occurs at a transfer time of seven minutes.

3.4.5.3 Tanker Vehicle Vent System

Two concepts were considered for tanker vehicle venting during the parking orbit (zero "g"). The first concept assumes that the vapor ullage bubble occupies the approximate center of the tank and is surrounded by the fluid in a zero "g" environment.

This approach considers forward venting with an aft extending stack that reaches the volume of the vapor bubble (optimum location of the vent stack aft end would be at the geometrical center of the tank). When pressure build up occurs due to heat input, the relief valve vents out the excess pressure. To insure proper operation in any condition, a slight modification to the standard type of venting system is required as shown in Figure 3-23. This modification consists of adding a double action valve which is controlled by an inertial sensing device. When acceleration is sensed, the valve is positioned such that flow through the short stack (normal ullage outlet) is permitted and flow through the extended stack is prevented. When zero "g" is indicated, the process will be reversed so as to minimize fluid loss.

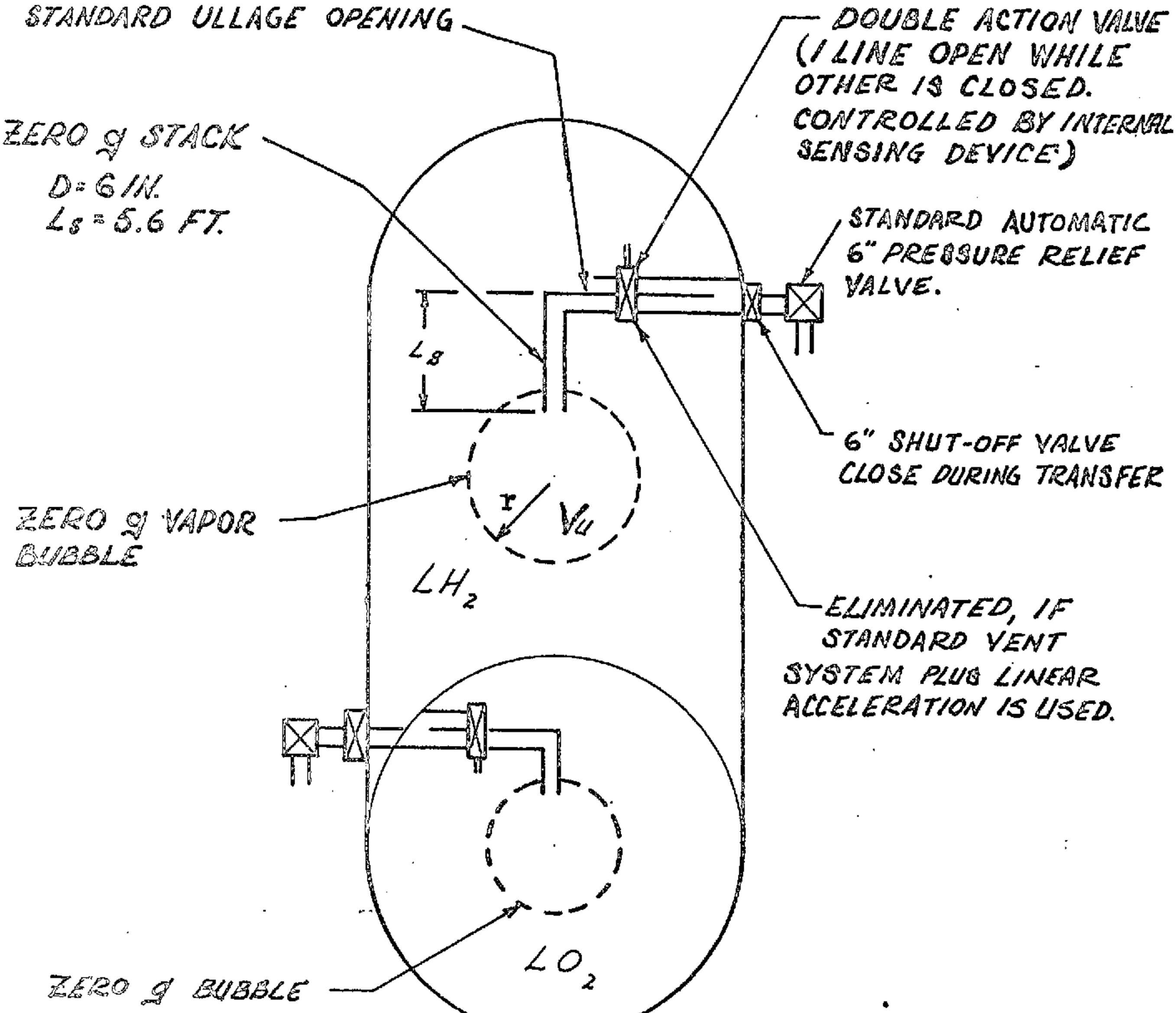
The second approach considers linear acceleration by the auxiliary propulsion subsystem at the time venting is required. This method requires additional equipment to control the auxiliary propulsion system for start and cutoff operations. However, it does not require the condition assumed for the first concept. Thus, if for some reason this condition does not occur, such as in the case of agitated storage or due to perturbing forces, the second approach will allow venting without dumping liquid. Approximately 20 pounds of propellant is expended for settling during each venting.

3.4.5.4 Orbital Launch Vehicle Vent System

The liquid hydrogen tank of the orbital launch vehicle booster has two separate vent systems for orbital operations as shown in Figure 3-24. The standard vehicle vent system is similar to that installed on the tanker vehicle and is used for venting under zero "g" conditions and as required during launch. The other system is designed to limit the maximum tank pressure during propellant transfer. During propellant phase separation, vapor collects at the aft end of the tank due to linear acceleration. Therefore, a separate provision is required to vent out pressures in excess of the established values. The transfer vent exhausts are arranged and designed so that they deliver additional thrust (approximately 42 pounds total) for linear acceleration.

COMBINED VENT SYSTEM FOR NORMAL AND FOR ZERO g CONDITION OPERATION

TANKER CAPACITY 200K LBS.



NOTE: DUCTING OMITTED FOR CLARITY.

FIGURE 3-23

COMBINED VENT SYSTEM FOR NORMAL AND ZERO g CONDITION OPERATION

OLV CAPACITY = 200K LBS.

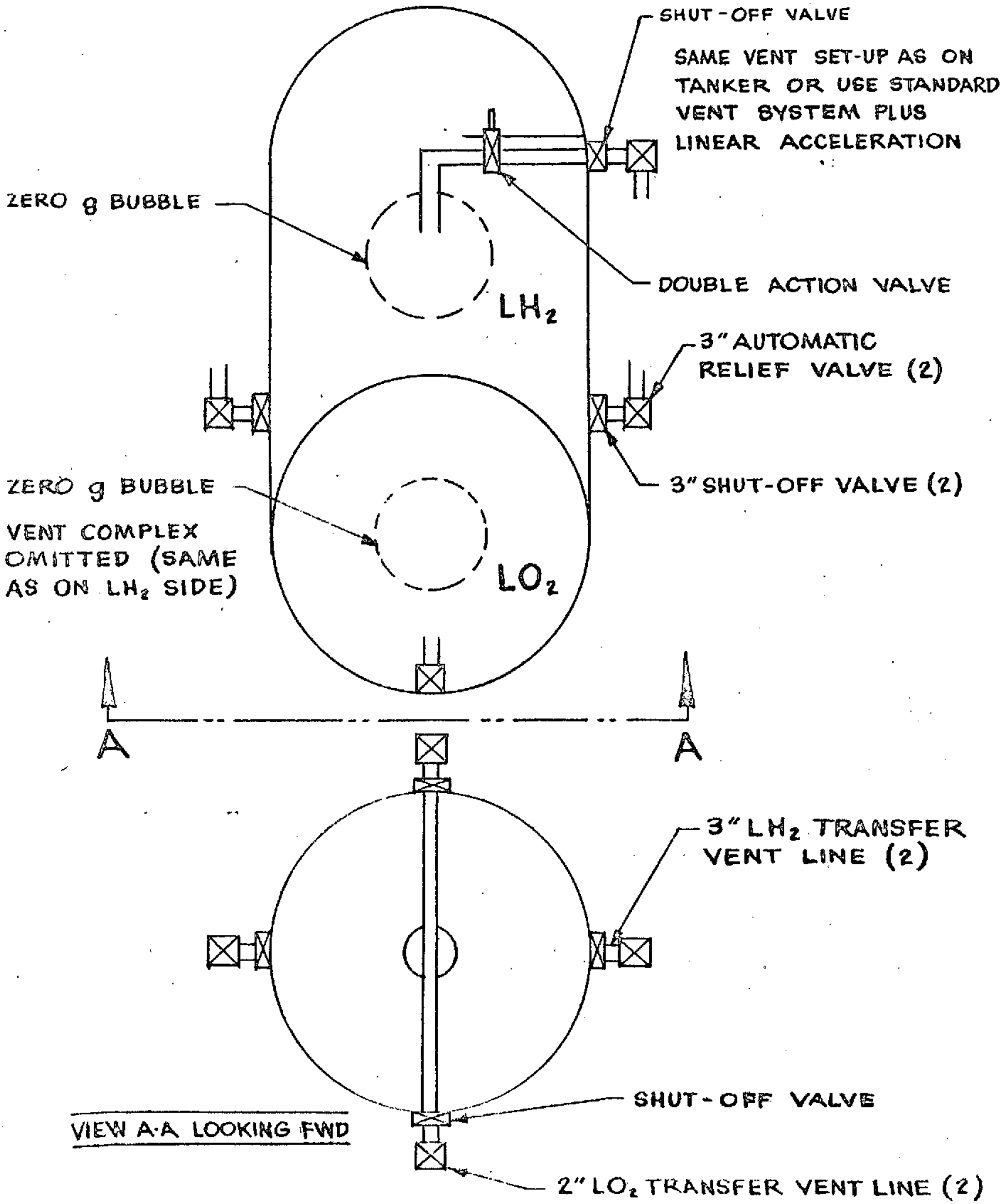


FIGURE 3-24

3.4.5.5 Auxiliary Propulsion Subsystem

The auxiliary propulsion subsystem installed on the tanker vehicle has a three-fold purpose. First, it produces linear acceleration for liquid-vapor phase separation prior to propellant transfer; second, it provides for tanker vehicle separation from the disengaged OLV after propellant transfer; and third, it generates occasional linear acceleration on command signal for venting the propellant tanks in the parking orbit. These units are protected by an insulated (jettisonable) aerodynamic fairing during atmospheric exit.

Design of the auxiliary propulsion subsystem was dictated by the propellant transfer time (10 minutes), the selected "g" level for propellant phase separation (10^{-3}g), and the combined weights of the tanker and the orbital launch vehicles (417,000 pounds). The resulting design parameters for a typical propulsion subsystem are listed below:

(a) Propellant

  (1) oxidizer - IRFNA (Density = 91.8 lb/ft3)

  (2) fuel - MMH (Density = 54.8 lb/ft3)

  (3) mixture ratio - 2.3:1

(b) Thrust - 225 pounds per operating nozzle (two of four nominal operation)

(c) Chamber pressure - 100 psia

(d) Expansion ratio - 30:1

(e) Vacuum specific impulse

  (1) steady state operation - 303 sec

  (2) pulsating operation - 290 sec

(f) Tank pressure - 150 psia

(g) Pressurant - helium

(h) Pressurant storage pressure - 3,000 psia

(i) Positive expulsion system - double teflon bladders

(j) Propellant and pressurant tank shapes - spherical

(k) Operating Time - 200K pounds of propellant transferred

(1) 620 seconds during transfer

(2) 60 seconds for venting operations

(3) 20 seconds reserve and separation

(4) 700 seconds total

(l) Propellant supply - 540 pounds per unit x 2 units = 1080 pounds total.

3.4.5.6 Propellant Transfer System Vehicle Arrangement

The general arrangement of the linear acceleration propellant transfer system is shown in Figure 3-25. The items directly associated with the system are labeled. The arrangement considers the docking method of attaching the tanker vehicle forward end to the OLV aft end. The major portion of the docking structure is carried by the tanker to allow engine operation of the OLV vehicle prior to orbital operations.

The tanker vehicle is shown without a main propulsion system, however, the design presented is compatible with the tanker vehicle equipment with a main propulsion system in the event this method is required to inject the tanker into orbit. Of necessity, however, the tanker vehicle must then include additional ducting, valves, pressure supply, thrust structure, etc., for a main propulsion system operation. Since the tanker vehicle is intended to accelerate in the direction of LO_2 tank leading during transfer operations and storage venting operations, a vent system must be installed for these ends of the LH_2 and LO_2 tanks, as well as the normal vent system at the other end of the tank (utilized during earth launch operations).

3.4.5.7 Propellant Mass Gaging

A means of determining the mass of propellants in the tanker vehicle under the zero 'g' environment of orbital storage and during the actual propellant transfer operation would be required for orbital propellant transfer operations. The system should be capable of operating accurately over a wide range of propellant tank temperatures and pressures. In addition to operating under zero 'g' conditions, the system should be capable of gaging propellant mass under micro 'g' and higher acceleration levels. The system should be capable of replying immediately to interrogation from ground stations and the spacecraft.

An electrical capacitive gaging system is proposed for this requirement. The system consists of paired capacitor plates (or wires) extending throughout the propellant tank. The propellant is considered to be in a liquid-vapor state. Because the liquid has a higher dielectric

ORBITAL REFUELING VEHICLE
GENERAL ARRANGEMENT

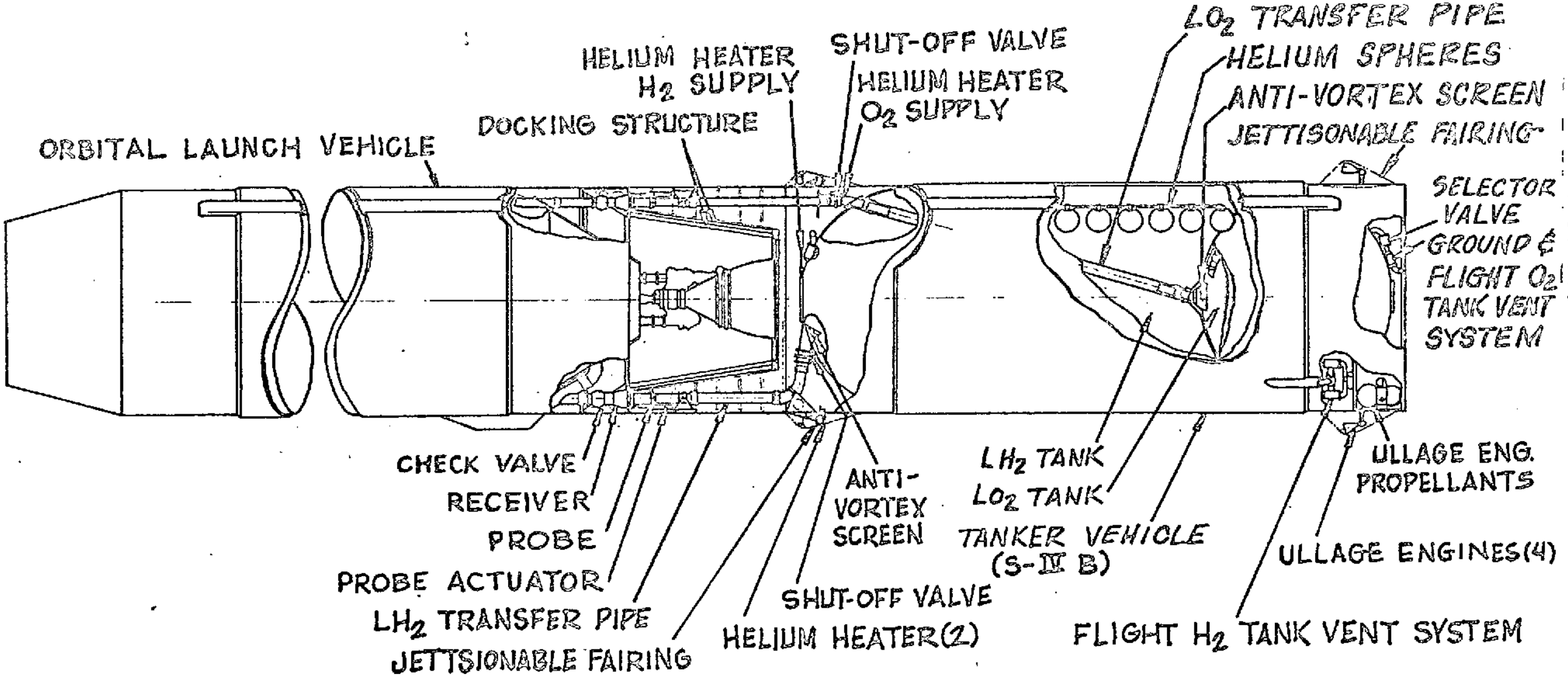


FIGURE 3-25

[redacted]

constant than the vapor it displaces, the capacitance of the sensor will be proportional to the volume of liquid in the tank. In addition, because the dielectric constants for hydrogen and oxygen are very nearly a linear function of their densities, this capacitance readout can be considered to be an indication of propellant mass.

Since the orientation of the propellant within the tank cannot be predicted under zero 'g' conditions, it is necessary to integrate the readout over the entire tank volume. The best method which has been suggested to do this is the Matrix Liquid Quantity Sensor developed by the Liquidometer Corporation of New York. This device uses a three-dimensional wire matrix to divide the sensor volume into a large number of capacitive cubes which are essentially paralleled. This device is heavy and complex. It is therefore proposed to incorporate in the tanker a simplified system using the same principle. This would make use of a number of wires stretched longitudinally in the tank. The wires would be electrically isolated from the structure and maintained under tension. They would be arranged in a grid pattern as viewed in a tank cross-section. By controlling the spacing of the wires, a near uniform field can be achieved throughout the tank. Twenty wire pairs in an S-IVB tank would provide adequate resolution for gaging to an accuracy of 2% to 5%. The weight of the wires and fittings is expected to be less than 50 lbs. per tank.

In addition, a specialized computer would be required to transform the capacitance measurement into a propellant mass measurement. For the case presented, with only twenty pairs of capacitors, this may be more efficiently performed by a computer at the interrogation station, from capacitance reading telemetered from the tanker.

3.4.5.8 Storable Propellants Transfer

The linear acceleration method of propellant transfer may also be utilized to transfer storable propellants. Vapor pressures of hypergolics at expected propellant temperatures (490°R) are comparable to recommended cryogenic propellants pressures. As a result, the propellant tank design pressures and the propellant transfer pressures are also comparable.

The pressurization systems selected for the cryogenic propellants transfer, however, are not suitable for the storable propellants. There is no convenient cryogenic environment to store cold helium, however, other methods of providing pressurizing gas are available. Because of storable propellant physical characteristics, current pressurization system designs employ a solid propellant charge for gas generation. This system is reliable, lighter, and less complex than the cryogenic-helium heater system. It cannot be employed with the cryogenic propellants case because of the higher gas temperatures resulting in excessive heat transfer to the cryogenic propellants.

Boiloff of propellants should not be a serious problem with storable propellants, however, the fluid control problem is complicated by the existance of a much higher freezing point in the temperature control range. It is possible that a heating system may be required for orbital storage or a precise heat transfer balance be maintained in the storage tanker design, transfer operation, and the receiving OLV.

An auxiliary propulsion system similar to the one suggested for use with the cryogenic system may be employed. This system may operate off of a bladder operated reservoir connected with the main storage tanks.

The same venting considerations apply to storable propellants with the exception that the propellants, being hypergolic, constitute a greater possible hazard if vented simultaneously and in near proximity in a confined region where dispersion is not essentially instantaneous. For a similar reason leakage may constitute a greater danger with storables.

Tankage, ducting, and sealing materials which are compatible with storable propellants will differ from those employed for cryogenic propellants.

Because of the higher densities, settling time at a given "g" level will probably be less for the storable propellants than required in the case of LO_2-LH_2 systems. In addition, it is expected that smaller masses of storable propellants will be transferred, further reducing the propulsion requirements during the transfer operation.

3.4.6 Reliability

The reliability of the major components of the linear acceleration transfer system was estimated and the system expected success probability determined. Estimates accounted for growth in the component reliabilities. Augmentation of system success probability by manual repair and operation was not included in the estimates.

The linear acceleration transfer method requires the use of four small (100 lb. thrust) storable hypergolic engines for providing the necessary acceleration. The system, however, is designed to operate with two engine out capability. In addition to the engines, two electrical actuators are employed to extend and retract the propellant transfer lines, and a set of helium bottles and a helium heater are used to pressurize the tanker tanks. As far as achieving orbit and completing the docking maneuver are concerned all six propellant transfer systems are considered equal.

The expected probability of the equipment mentioned above operating successfully for the required period of time is in the range of 0.94 to 0.976. It is conceivable that this could increase appreciably as the items are better defined and tested. Table 3-3 shows the predicted component reliabilities.

	INITIAL (4th Quarter 1965) Individual	System	GROWTH (Early 1967) Individual	System
Engines	0.96	0.99	0.99	0.999
Actuators	0.99	0.98	0.995	0.990
Helium Heater		0.98		0.990
Other Equipment		0.99		0.995
Total System		0.94		0.976

TABLE 3-3 COMPONENT AND SYSTEM PREDICTED RELIABILITIES

The results presented in Table 3-3 were used in combination with a typical OLO program (Initial lunar landing in April 1967, utilizing the Saturn C-4 and the conventional tri-propulsion Apollo spacecraft.) to obtain the probability of successful propellant transfer operations. This is shown in Figures 3-26 and 3-27 as a function of the operations dates for both the lunar orbit and lunar landing missions. The success probability of the propellant transfer operation itself is greatly enhanced by the addition of a spare tanker in the system. However, the additional docking operation required in this event may negate this enhancement, depending upon the success probability of the docking operation.

Manual repair, or operation, would also improve the success probability; however, the magnitude of this effect should not be evaluated until man's capability in orbital operations external to a protected environment has been further verified.

The linear acceleration system may be considered independent of an OLF as far as operational reliability is concerned. This is because redundancy is included in the propulsion system and could be easily included in the heater system as well (the two major components of the system), thus the ability of the OLF to replace and repair components contributes little to the system reliability. Also, during operation, the tanker in the linear acceleration system would be removed from the vicinity of the OLF and thus cannot be augmented by equipment on board the OLF. The simplicity and straightforwardness of the linear acceleration approach, compared to other transfer concepts, appears to offer the highest probability of successful operation, with or without an OLF.

3.4.7 Development Program

3.4.7.1 System Component Development

Figure 3-28 lists the major components of the linear acceleration propellant transfer system along with their development schedule. In addition,

PROPELLANT TRANSFER PHASE,
LINEAR ACCELERATION SYSTEM

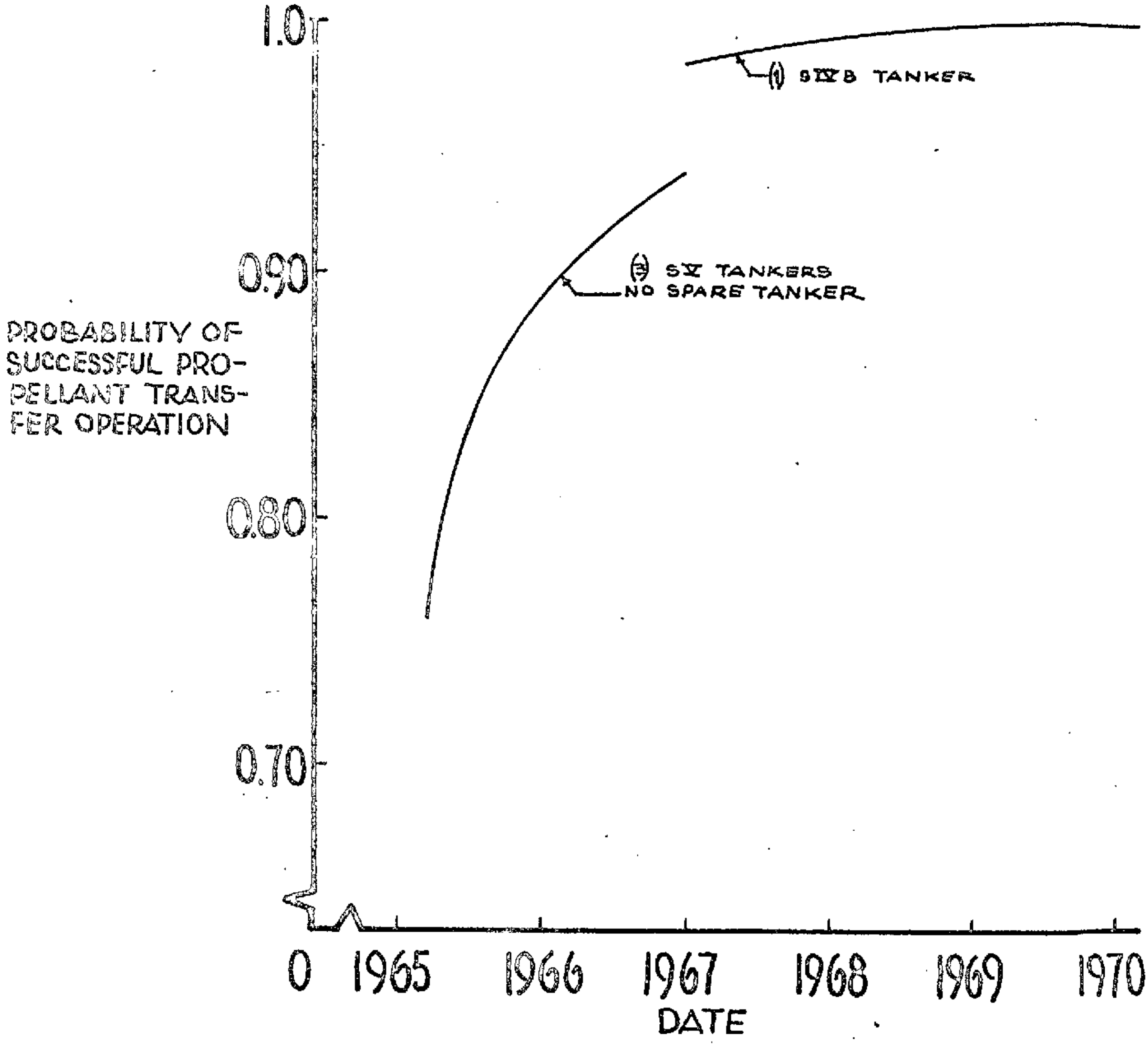


FIGURE 3-26

PROPELLANT TRANSFER PHASE,
LINEAR ACCELERATION SYSTEM

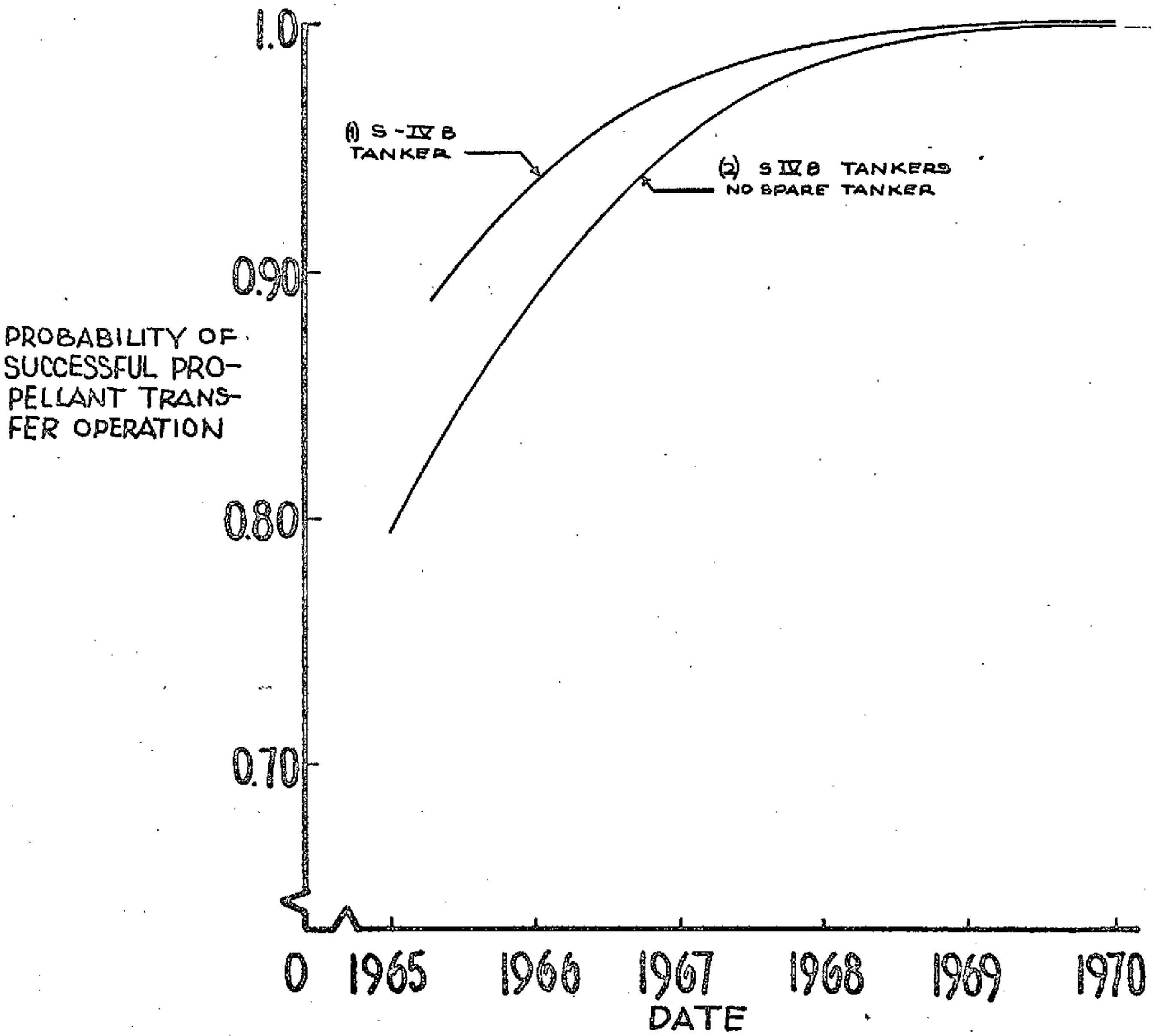


FIGURE 3-27

LINEAR ACCELERATION DEVELOPMENT PROGRAMS

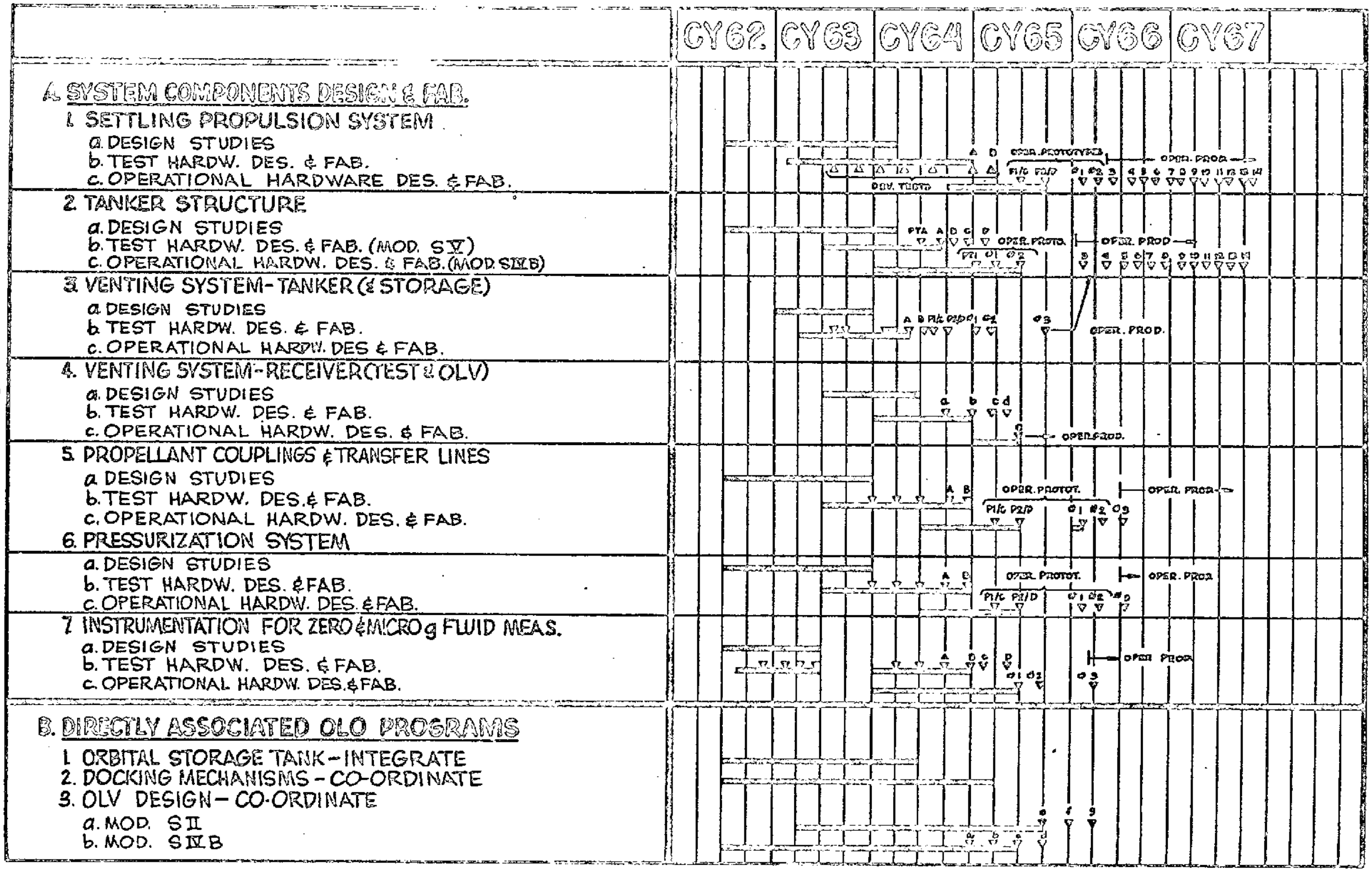


FIGURE 3-28

the directly associated OLO programs are shown with their development coordinated with the propellant transfer system. The development program is scheduled for an operational system in April, 1967, the projected first manned lunar landing mission orbital launch as specified by the OLO program used as a guideline. In this program the operational tanker is a modified S-IVB. For an accelerated OLO program the operational date would be May, 1965 for the S-V tanker and September, 1966 for the S-IVB tanker. This would require some compression of the development schedule and the development of the S-V as an operational tanker as well as the S-IVB. Since the S-V is slated to be utilized as a test vehicle in the nominal program, this additional requirement can be easily phased into the program. Test vehicles A, B, C, and D are modified S-V's. Units #1 and #2 are prototype operational systems for use on the first two S-IVB tankers. The first operational S-IVB tanker is Unit #3. In addition, some subsystems are tested in prototype form (P_1 and P_2) on the S-V Test Vehicles C and D.

The production schedule anticipates three operational orbit launches in 1967 and six in 1968, each mission requires two S-IVB tankers.

3.4.7.2 Test Program

The test program is shown in Figure 3-29 along with the booster requirements for test launches and the delivery and launches of the operational tankers. An important requirement for the development program is early investigation and tests of the behavior of LO_2 and LH_2 under zero and micro g conditions, and the development of the necessary instrumentation for these tests. While some tests are possible with aircraft flying special trajectories, these are not considered satisfactory for two reasons: first, the trajectories do not attain a true zero or even micro g conditions due to the tolerances and control responses required (trajectories of this type achieve a condition of $0 \pm .05$ g as an average); and second, the duration of these conditions are very brief, on the order of seconds, and are not considered truly representative of orbital conditions. Since analysis indicate that accelerations as low as 10^{-5} g have a profound effect on the fluid behavior, only high altitude ballistic flights or orbital flights provide the necessary conditions. For these reasons it is desirable to provide orbital tests of the zero g research tanks. These are expected to weigh on the order of 1,000 pounds and may be placed into a 300 nautical mile orbit with a Thorad A/Agena B booster in 1963.

In addition to orbital tests on fluid behavior, subsystems of the propellant transfer system should be tested in orbit. These will utilize an 8,000 pound payload boosted by Atlas/Centaur and a modified S-V tanker system loaded to 20,000 pounds propellant and boosted into orbit by a C-1B vehicle.

Ground tests include engine firing, vacuum, and orbital simulator environment tests. These will utilize existing facilities.

3.4.7.3 Costs

The development cost and costs for the first six operational units (first year (1967) of orbit launch operations for nominal OLO program) are

LINEAR ACCELERATION DEVELOPMENT PROGRAMS

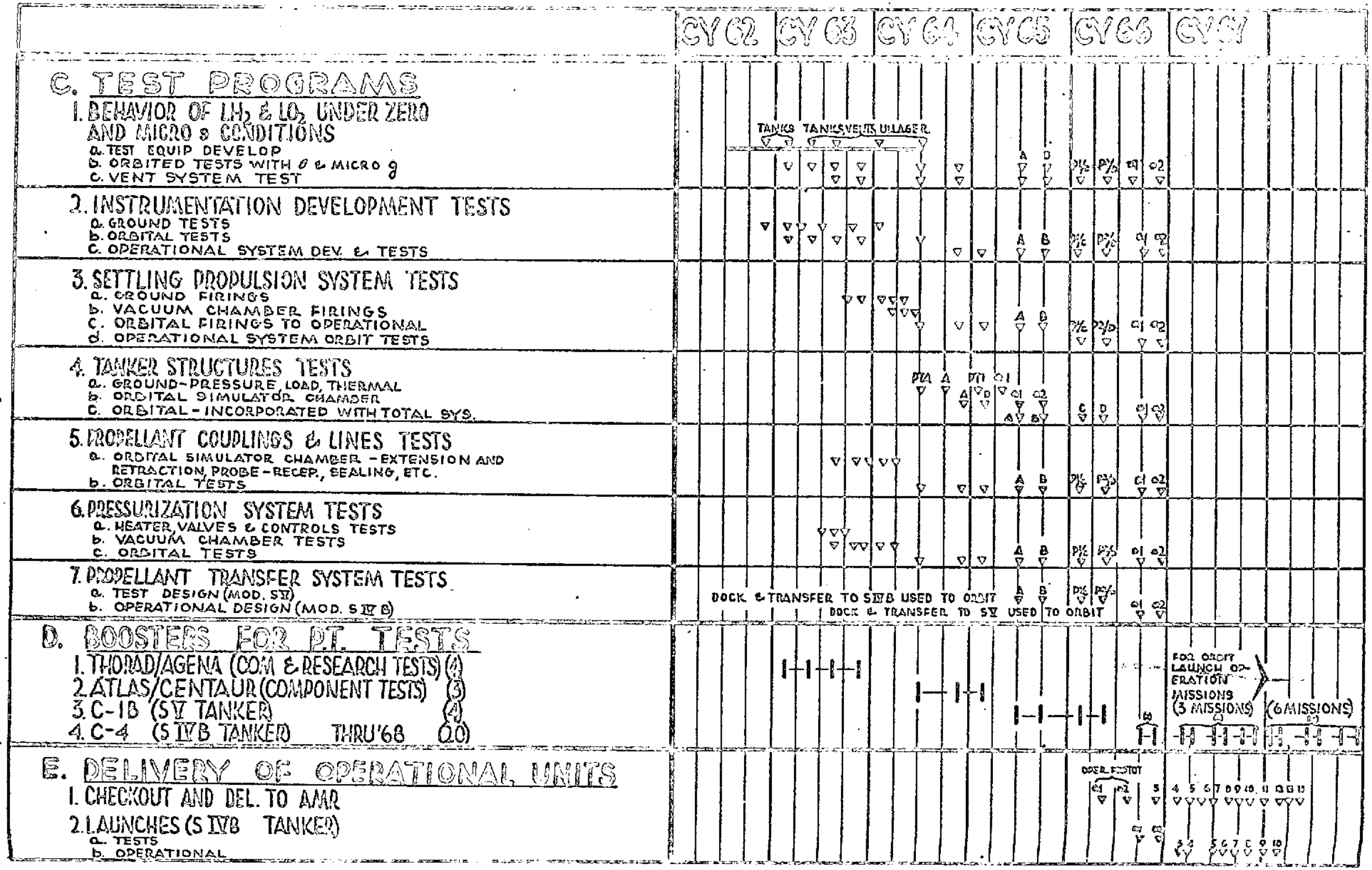


FIGURE 3-29

CONFIDENTIAL

shown in Table 3-4. The breakdown is for fiscal years 1963 through 1967 and including three operational tanker launches in FY 1968. Cost of booster and launches are not included, although the Thorad A/Agena B booster costs are included separately and the total number of each type of booster utilized are noted.

Costs are noted as incurred.

TABLE 3-4

COSTS OF DEVELOPMENT AND FIRST YEAR OPERATION OF LINEAR ACCELERATION PROPELLANT TRANSFER PROGRAM

Item	Cost - Millions of Dollars						
I. Non Recurring	Fiscal Year						
	63	64	65	66	67	68	Total
1. Engineering Studies	2.1	1.9	1.0	0.8	0.3	0.1	6.2
2. Tanks and Systems (2 mod. S-IVB plus equip.)		5.0	15.0	25.0	5.0		50.0
3. Test Tanks and Instr. (zero g tests)		1.9	1.0	2.0	0.4		5.3
4. S-V Tanks and Mod. (Prototype System Tests)		1.0	1.0				2.0
	2.1	9.8	18.0	27.8	5.7	0.1	63.5
5. Boosters*(tests)							
Thorad/Agena (4)	(18.0)	(6.0)					
Atlas Centaur (3)							
C 1B (4)							
C 4 (2)							
II. Recurring							
1. Tanker (6 mod. S-IVB)			4.0	16.2	11.2	1.0	32.4
2. Launch Operations (Tanker only)					1.0	0.6	1.6
3. Transportation					1.8	0.7	2.5
			4.0	16.2	14.0	2.3	100.0
4. Boosters* C-4 (6)	*Booster Costs are Excluded						

[redacted]

3.4.8 Growth Potential

The linear acceleration system is adaptable to either moderate (20,000 to 100,000 lbs.) or large (100,000 - 400,000 lb.) propellant loadings. The linear acceleration hardware may develop into three or four basic modification kits for each vehicle size by simply increasing the propulsion system and adding pressurization supply bottles. It tends to minimize the changes required in adapting any developed vehicle to orbital propellant transfer tanker or OLV. Many of the subsystems developed for its operation, such as the pressurization system and propellant couplings, may be adapted to the rotating assembly method if this method is employed in advanced versions. This may be the case for an OLF system with a large rotating "fuel dump."

However, unless experience proves otherwise, the linear acceleration system appears to be quite adaptable to an OLF system since the orbit change produced by the transfer operation is minor and can be programmed to return to the OLF without additional propulsion requirements by breaking the transfer operation into two phases.

APPENDIX A

Docking Method Considerations

A.1 Comparison of Rigid and Non-Rigid Attachment

General docking concepts were considered for the 230,000 pound orbital transfer tanker (based on S-IV type of vehicle). Rigid attachment between the tanker and the OLV appeared to be more advantageous than non-rigid attachment for the following reasons:

a. Automatic hook-up is more feasible.

b. Attitude control sequencing will be a minimum during the transfer operation.

c. Attitude control of the two vehicles during transfer operations would be accomplished as a single vehicle (minimizing the risk of parting transfer lines). Also, the control mode would be less complicated than controlling two vehicles separately and with respect to each other while transfering propellants.

d. Propellant transfer lines would not be used as structural members.

e. For the linear acceleration method of propellant transfer an auxiliary propulsion system is required for operation on both tanker and OLV.

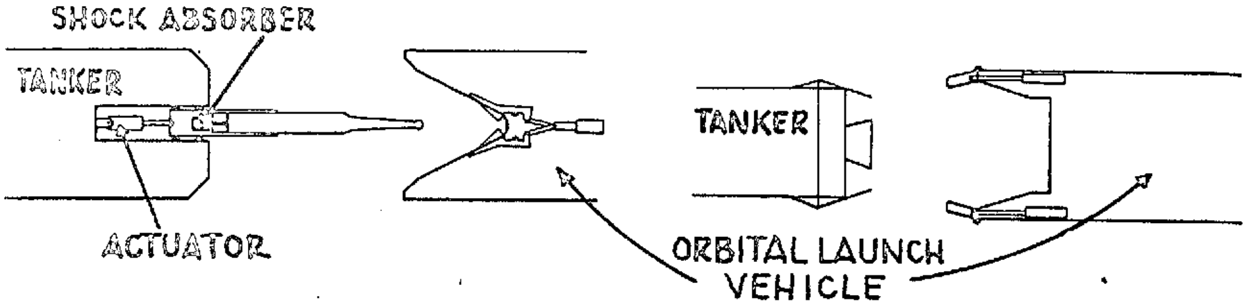
A.2 Vehicle Docking Modes

Three rigid docking schemes were considered; end-to-end, side-to-side, and side-to-end. Figure 3A-1 shows various methods for end-to-end docking arrangements. Each system has a self-alignment feature for docking. Various hook-up and shock absorbing devices are shown. Shock absorption would be accomplished by damped springs, oleo-pneumatic devices, or gas bags. Honeycomb shock absorption systems cannot be employed since the energy is absorbed by crushing the honeycomb and is therefore good for only one docking maneuver. One illustration of a non-rigid end-to-end docking arrangement is given for comparison.

Side-to-side coupling methods are shown in Figure 3A-2 for a rigid and a non-rigid attachment. Control of the docking and energy absorption problems are more severe in this mode of attachment.

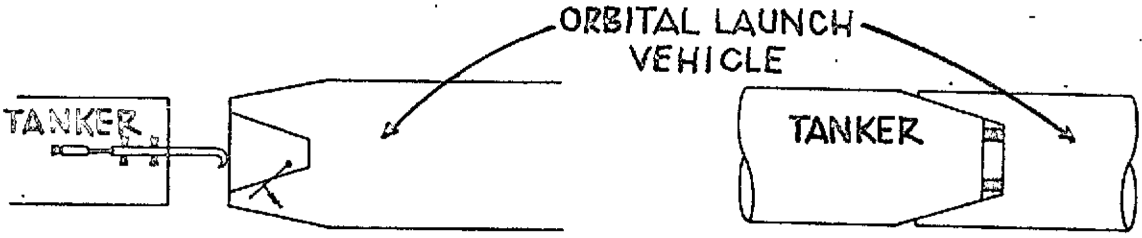
Figure 3A-3 illustrates side-to-end docking arrangements. In this docking mode automatic hook-up of propellant lines becomes more complex as well as the increased complexity of the docking operation and mechanism.

END TO END DOCKING



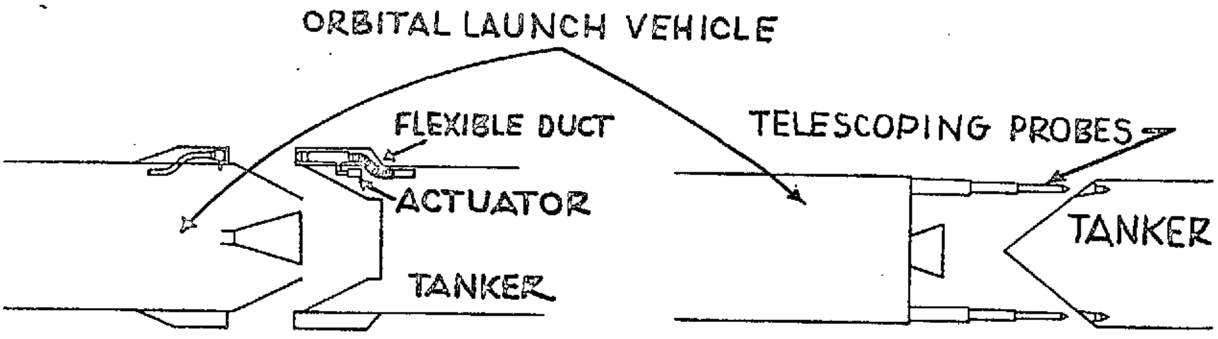
WITH TELESCOPING PROBE

MODIFICATION OF MATING CONES



WITH RETRACTING HOOK

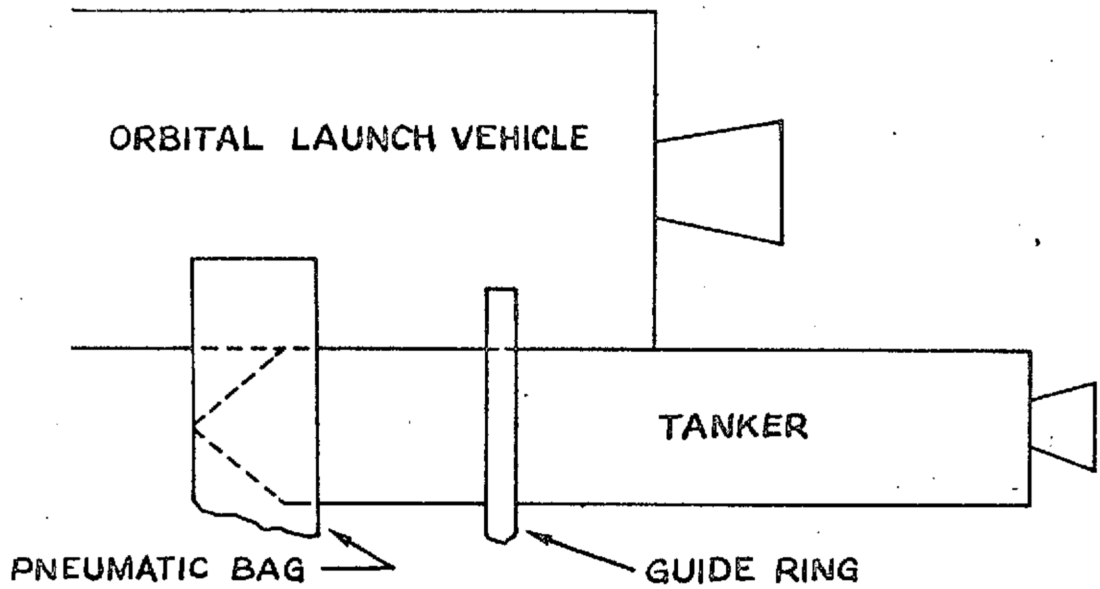
WITH MATING CONES



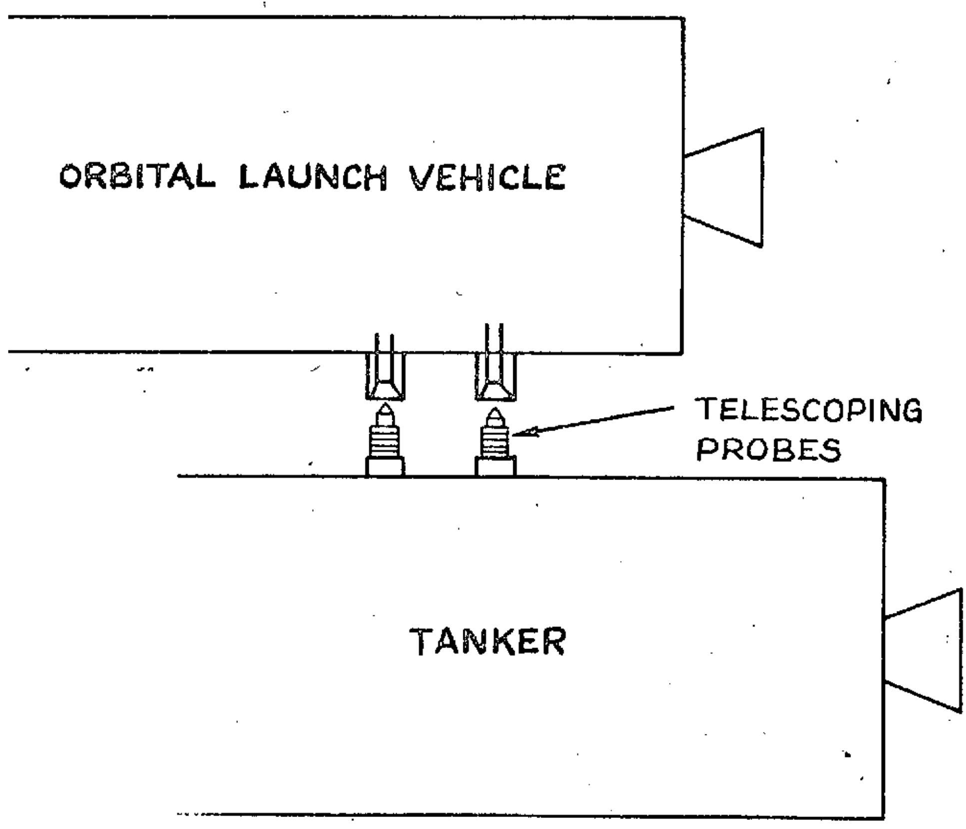
WITH EXTEND-RETRACT TRANSFER COUPLING

NON RIGID WITH TELESCOPING TRANSFR PROBES

FIGURE 3A-1

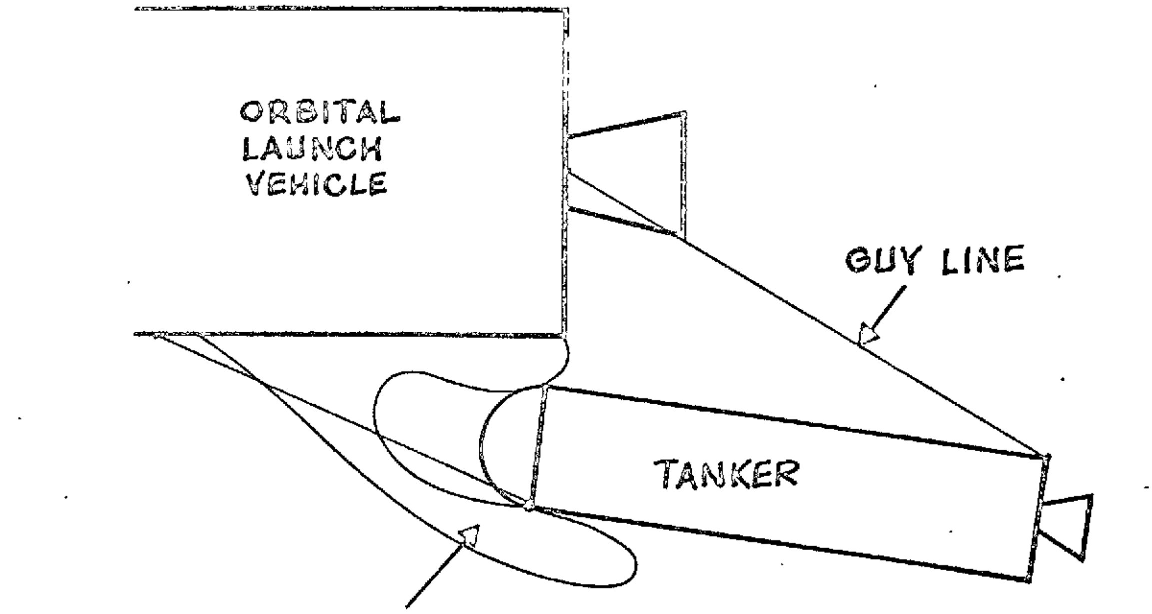


METHOD WITH PNEUMATIC
BAG AND GUIDE RING

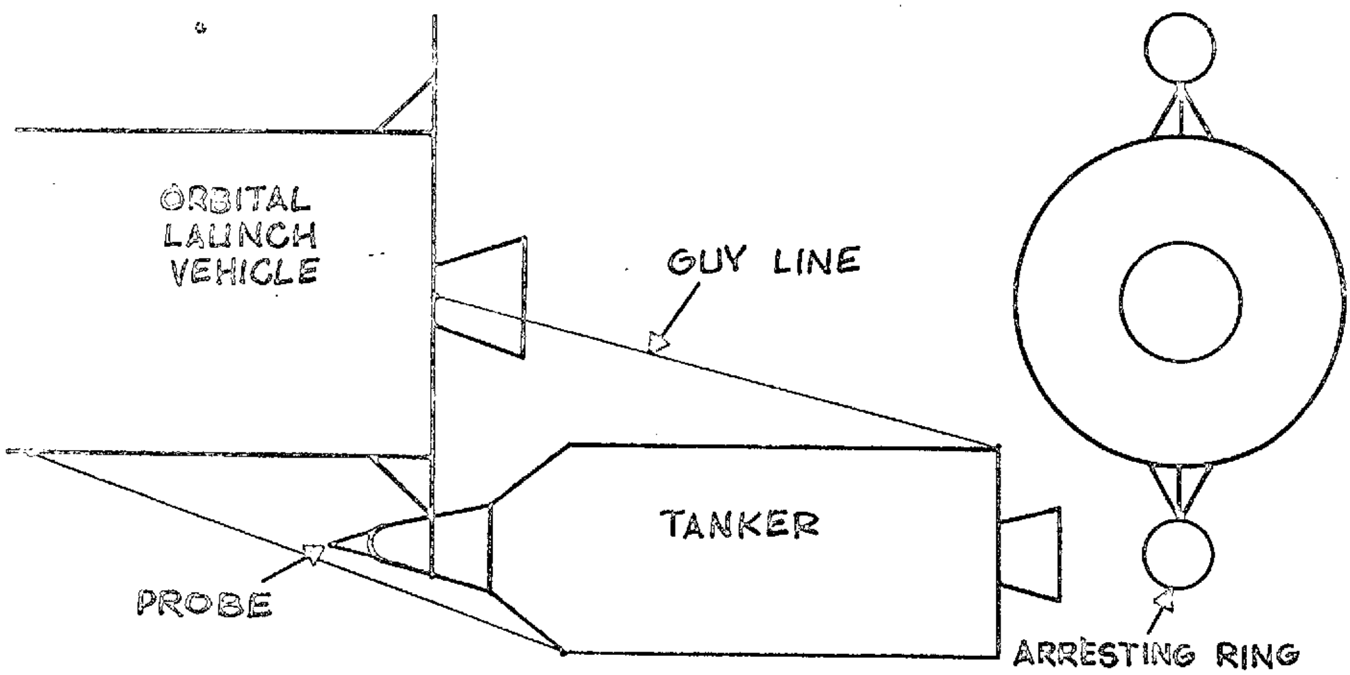


NON RIGID

FIGURE 3A-2



WITH PNEUMATIC BAG AND GUY LINES



USING PROBE AND ARRESTING RING

FIGURE 3A-3

The end-to-end concept appears most feasible for the following reasons:

a. The vehicles can withstand higher static and dynamic loads in the longitudinal direction than in the transverse directions.

b. Longitudinal loads introduce less beam bending to the vehicle.

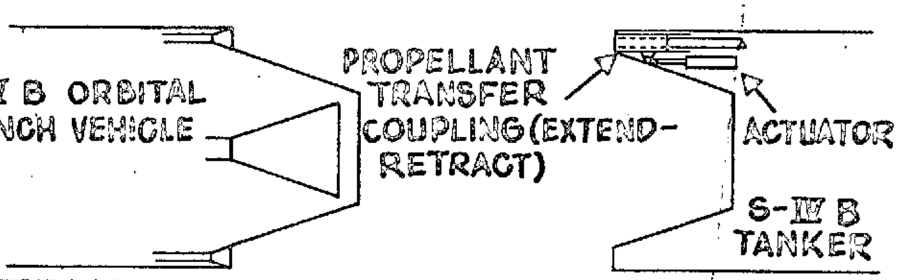
c. Longitudinal loads introduce smaller disturbing moments to be corrected by the attitude control system.

d. Automatic hook up is more applicable to the end-to-end arrangement.

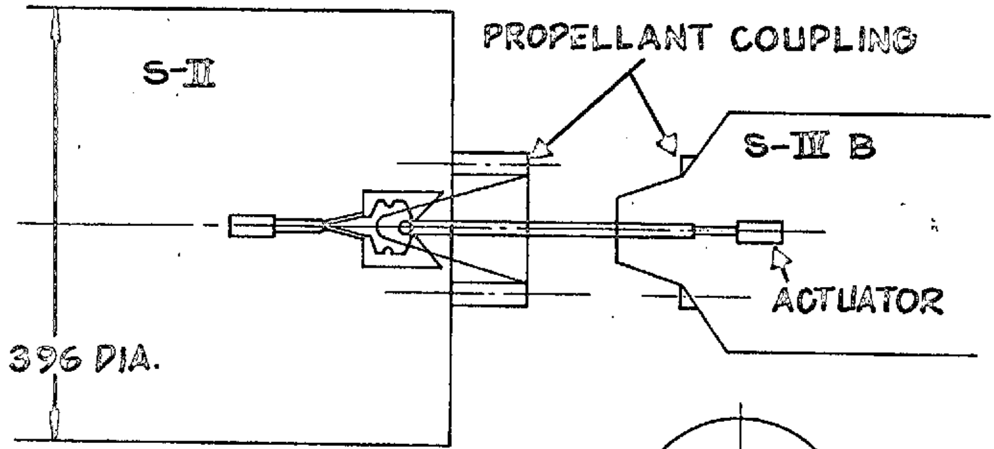
A.3 Docking of OIO Propellant Transfer Vehicles

The docking arrangement between combinations of OLV and tanker vehicles for orbital propellant transfer operations are shown in Figure 3A-4 . Mating cones are employed with an assist from a retracting probe and claw mechanism where, during the terminal phase of docking, the OLV engines are not protected by docking structure. All methods employ an end-to-end attachment made with propellant couplings retracted during docking and extended into the receiving drogue to accomplish propellant transfer after completion of the vehicle assembly.

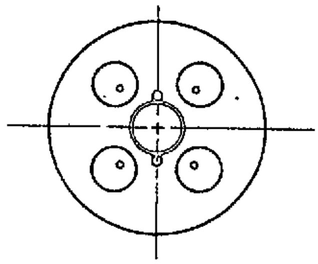
END TO END DOCKING

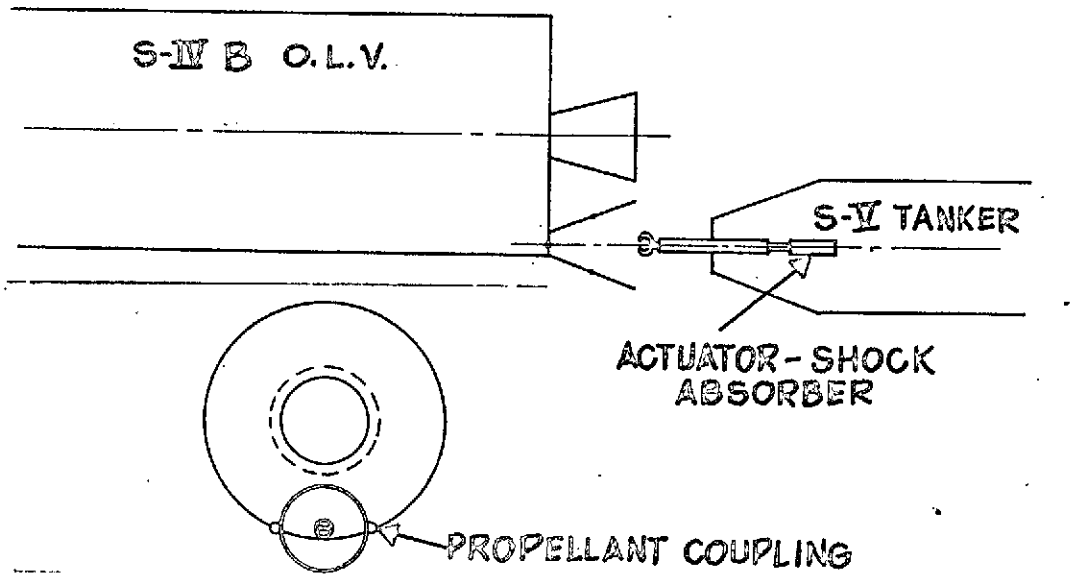


WITH EXTEND-RETRACT TRANSFER COUPLING



PROBE AND CLAW WITH
MATING CONES





ATTACHMENT PROBE AND CLAW WITH MATING CONES

FIGURE 3A-4

APPENDIX B

B.1 Propellant Losses

Propellant losses are due to heat transfer into the liquid hydrogen propellant tank from the following three sources (LO_2 losses are small compared to LH_2): (1) aerodynamic heating during earth atmospheric exit, (2) solar radiation, plus earth albedo and infra-red during orbiting, and (3) heat leak through the common bulkhead from LO_2 (OLV only). Approximately 160,000 BTU (S-IVB) are generated by aerodynamic heating. To minimize propellant vaporization during transfer, the OLV tank should be maintained at liquid hydrogen temperatures prior to fill. For this example this can be attained by partially filling the fuel tank with 5,000 pounds of liquid hydrogen to absorb heat input from all sources.

During orbiting, the OLV (standard insulation) absorbs 166,000 BTU per day. This heat is absorbed by approximately 900 pounds of liquid hydrogen (stratified - without agitation). Assuming that the OLV orbits for three days after propellant transfer, the propellant loss due to orbiting is 2,700 pounds.

The liquid hydrogen absorbs 72,000 BTU per day through the common bulkhead from the liquid oxygen tank. This heat leak results in a fuel loss of 400 pounds per day. The total propellant boil-off losses of the OLV for orbital periods of one, two, and three days respectively are summarized below:

	1 day	2 days	3 days
Aerodynamic heating	850 lbs.	850 lbs.	850 lbs.
Orbital heating	900 lbs.	1,800 lbs.	2,700 lbs.
Bulkhead heating	383 lbs.	766 lbs.	1,149 lbs.
TOTAL LOSSES	2,133 lbs.	3,416 lbs.	4,699 lbs.

The above tabulation suggests that the OLV should be limited to a maximum orbital period of one day to minimize propellant loss.

The fuel tank of the tanker vehicle utilizes a more efficient type of insulation (approaching super insulation) which reduces heat transmission to the tank to 17,000 BTU per day. Thus, for a 30-day storage period, 2,700 pounds of liquid hydrogen are lost. The total hydrogen boil-off due to aerodynamic heating and a 30-day storage period is 3,550 pounds.

[redacted]

PART IV
CREW TRANSFER

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1.0 CREW TRANSFER

1.1 INTRODUCTION

The orbital launch approach to the accomplishment of the Apollo Manned Lunar Landing mission has been proposed. This approach may be accomplished using either of two techniques; Orbital Assembly or Propellant Transfer.

Crew transfer can be used in conjunction with either of these techniques to effect an earlier manned lunar landing and to improve crew safety during the launch into earth orbit phase of the orbital launch operation.

Crew transfer is the process of moving the crew from one vehicle to another in space.

1.2 DISCUSSION

In order to accomplish an orbital launch the space vehicle or the component parts thereof, must be put into orbit first, then, either assembled or refueled, and launched into lunar trajectory.

One of the prime advantages to the orbital launch approach for the initial manned lunar missions is that smaller earth launch vehicles, some of which are existing, may be utilized. In order that the number of ELV's, required to inject the various components of the space vehicle into orbit, may be held to a minimum, moderately large ELV payload capabilities are required. At the present time, ELV's with the desired payload capabilities are either in the planning and design stage or have but one research and development launch to their credit.

It is essential that the reliability of the ELV which transports the crew to earth orbit be of the highest degree in the interests of crew safety. In order not to delay the initial manned lunar mission program while waiting for the Saturn C-4 to attain the desired reliability, the technique of crew transfer may be used to advantage. By taking advantage of the crew transfer technique, the space vehicle may be injected into earth orbit unmanned, by a Saturn C-4 while the crew may be transported on top of a Saturn C-1 or C-1B from which they would execute an orbital transfer after rendezvous.

By this technique, the Saturn C-4's would not be involved in crew safety and hence vehicles in the R and D program could be used. By the time period anticipated for the initial manned lunar missions, the Saturn C-1 will have participated in a sufficient number of launches to be well past the R and D phase and be operational with an attendantly higher reliability and confidence level.

In addition to the advantages in crew safety derived from this technique, possible advantages in either schedule acceleration or contingency against schedule slippage may accrue. During the subject time period, the launch rate schedules of the Saturn C-4 are assumed to be such that the utilization of R and D launch vehicles in conjunction with the crew transfer technique indicates a possible acceleration in the program schedule by as much as six months. The crew transfer technique may also be used as a contingency factor to absorb schedule slippages which may occur due to a combination of Saturn development problems and the published time interval between Saturn C-4 launches.

The benefits derived through the use of the crew transfer technique will be realized only until the launch rate schedules and reliability of the Saturn C-1 and C-4 coincide. It must then be recognized that a decrease in the probability of mission success is inherent in the crew transfer technique.

This decrease in the probability of mission success is axiomatic in that the overall probability is the product of the various component reliabilities. When crew transfer is used, this product must include the additional reliability factors associated with the Saturn C-1, rendezvous and crew transfer itself. None of these additional factors which must be included in the equation can, by definition, have a reliability of unity, hence the product must be less. Hence, an operation involving crew transfer (with a Saturn C-1B) plus an orbital assembly of two unmanned C-4 payloads has a lower mission success probability than an operation involving orbital assembly of two C-4 payloads, one of which is manned.

An economical advantage would be gained from the use of crew transfer during a later period of time. When travel to and from the moon, for example, had become commonplace, an economical means of travel would be to have large space vehicle shuttles operating from an orbit around the earth to an orbit around the moon, then have small ferry vehicles to transport personnel between the shuttle and the surface at both ends of the line. A crew transfer would be a part of this operation between the ferry and the shuttle.

The following is a discussion of techniques and considerations for utilizing crew transfer during orbital assembly as an example.

1.2.1 Orbital Assembly Utilizing Crew Transfer

When crew transfer is used with the orbital assembly technique, sufficient launchings must be made to insure that one of each segment required to assemble the space vehicle, is available in orbit. The crew would then be launched into orbit in a transfer vehicle to assemble the space vehicle and conduct the mission.

The crew would utilize the transfer vehicle as living quarters while assembling the space vehicle. It must therefore be sized to accommodate the crew for this period of time. The transfer vehicle would have to be placed within close proximity of the space vehicle or have the capability onboard of maneuvering into such a position after being placed in orbit.

Once in position in orbit the segments of the space vehicle would be brought together remotely by controls within the transfer vehicle, or directly by the crew leaving the transfer vehicle and operating controls on each space vehicle segment.

1.2.1.1 Remote Controlled Assembly

If the assembly is remotely controlled from within the transfer vehicle it is possible that the crew would never be exposed to the space environment. Once the space vehicle was assembled the systems could be tested by remote control and telemetry. When it is ascertained that the Command Module is safe for habitation the transfer vehicle would dock with the Command Module and the crew would transfer into the Command Module through air locks in each vehicle. Once this is done the transfer vehicle would be abandoned.

This technique would require that remote assembly controls and test equipment be a part of the transfer vehicle. No omni-environmental pressure suit would be required except in case of emergency or failure during joining. Grappling devices and maneuvering energy could be stored in each segment and the joining accomplished remotely by visual contact. An air lock would not be necessary in the transfer vehicle for cocking and crew transfer but would be required if a crewman had to go outside the transfer vehicle for any reason.

1.2.1.2 Direct Controlled Assembly

If the assembly is controlled directly by the crewman, outside the transfer vehicle, there would be no need for docking the transfer vehicle with the Command Module. In this case, however, a "space suit" of some sort would be required which would allow the crewman to move about and operate controls in the space environment. This "space suit" would protect the crewman against absolute vacuum, possible solar flare radiation, temperature extremes due to deep space on one side and solar radiation on the other, extremely high intensity illumination, meteorite impingement, etc. An air lock would be required in the transfer vehicle for exit and entry.

The space vehicle segments could be designed to include joining devices and controls for effecting the moving, aligning and joining of the parts. A unit would be required to allow the crewman to maneuver and travel short distances once he was outside the launch vehicle.

Test equipment would be required on board the transfer vehicle or in some of the space vehicle segments in order for the space vehicle to be operationally tested prior to launch into a lunar trajectory.

Intercommunication between the crewmen and between the inside and outside of the transfer vehicle will be required during transition.

Safety devices will be necessary to insure that a man, having had an emergency in space, can be recovered. They may be safety lines or emergency recovery vehicles, small heat seeking devices to direct and attach life lines, or something of this nature.

The crewmen would locate all parts visually or by radar. The parts would be gathered together to within the immediate area of the launch vehicle.

Once the parts are in the same area the crewman will control the assembly of the space vehicle. Probably only two of the crew of three will ever be outside at one time, always leaving one man inside in case trouble develops somewhere. Once the space vehicle is assembled the crew would check out all the systems either remotely or by the use of portable test equipment from the outside. Once this is accomplished the crew will abandon the transfer vehicle and any other equipment not necessary for the lunar landing mission.

1.2.1.3 Earth Orbital Mission

The Apollo earth orbital mission may require a laboratory as a part of the ELV payload. With the laboratory a part of the ELV payload the command module is required to have the capability of docking with the laboratory. There would be no requirement for "space suits" except in case of an emergency which would require outside repair. It would appear that under normal conditions this mission could be accomplished without the need for "space suits" or tugs or self maneuvering units if the rendezvous and semi-automatic docking capability is designed into the command module.

1.2.1.4 Air Lock

In any situation where a crewman must move from the inside of a crew compartment to the outside of the vehicle in space, an "Air lock" is necessary unless the crew compartment equipment is constructed to withstand periodic exposure to a vacuum.

This air lock must be integrated into the structure of the vehicle so that it will not cause excessive leakage which would unduly penalize the environmental control system. A telescoping air lock proposed in the Apollo competition helped relieve the limited space problem. It should be sized to accommodate a crewman in a pressure suit (hard or soft depending upon which is required) and whatever equipment he may require to have with him; environmental and maneuvering back-pack, tools, test equipment, etc. He will most likely have to bend over or completely turn around while in the air lock in order to close off one end and open the other end, or he will have to depend upon powered controls and indicators or help from some other crewman. Manual operation of all critical items would be desirable from a reliability and confidence standpoint. If the outside pressure suit is a hard suit this suit itself might form the airlock or air locks. In this case the suit would be designed to dock with the vehicle in such a way that a seal would be accomplished around the entrance hatch and the entrance hatch would open into the vehicle.

The controls for the air lock entrance hatch should be of a type which would allow the crewman to brace himself properly when operating them. A rotating motion on the inside door, as a submarine hatch, would be difficult to operate because of the difficulty in bracing the body against rotation in the absence of gravity. A lever which could be pulled would allow bracing to counteract the pulling force. Hand holds may be required inside the air lock to aid in maneuvering after entry. Similar problems would be encountered for actuating the inner door from inside and with the outer door. There may be a need to operate the air lock rapidly during emergency; therefore ease and rapidity of operation should definitely be considered. The rescue situation should also be considered where one crewman might be unconscious and would need to be accompanied by another crewman. If everything is manual and a crewman is inside the vehicle when the rescue is effected there would be no problem. If no one was inside to operate controls a rescue would be impossible.

Operating the air lock for the docking maneuver will require that a seal be made, between the docking vehicle and the vehicle to be docked, around the outside opening of the air lock. The outer door would have to be designed to operate within the opening into the joined vehicle.

The procedure for using the air lock would be: The crewman will don his pressure suit and gather together the equipment he is to use. Extend the air lock into the cabin, (if telescoping). Open the inner door. Crawl into the air lock with all equipment. Close inner door and assure seal. Maneuver until the outer door latching mechanism can be reached. Depressurize air lock (valve controls inside air lock) open outer door and exit. Close outer door and assure a seal.

When the air lock is used after docking it would not be necessary, except possibly as an emergency precaution, to wear a pressure suit while moving from one vehicle to the other, once it has been determined that a good seal has been made and can be maintained. This would be assured by physically tying the two vehicles together to prevent their drifting apart.

Consideration should be given to the use of a flexible interconnect between the two vehicles instead of docking as described before. The initial contact of the interconnect with the other vehicle would be a problem, however, it might not be as much of a problem as the docking of the vehicle. The flexible interconnect could be flown or guided into place by the crewman visually through his view port.

The problems associated with this method of transfer appear to be about the same as docking except the connection problem as described above and the relative motion problem would not be as great. The chance of blowout or meteorite impingement would be greater since the interconnect material would be flexible.

1.2.1.5 Self Maneuvering Unit

In order to accomplish an assembly in orbit where the crewman must have the capability of maneuvering and traveling short distances in space, Vought Astronautics is currently developing a Self Maneuvering Unit, under contract with the Air Force*, which will provide the crewman with this capability. The unit allows the occupant to maneuver in three axes, roll, pitch and yaw, and translate in two axes, fore and aft and up and down. The studies to date have considered orbital transfer range capabilities of up to five miles. A typical operation of the unit is as follows:

A crewman wishing to translate from one object to another in space utilizing the Self Maneuvering Unit (S.M.U.) would first orient himself such that the object he wishes to go to is directly in front of him. He then accelerates forward. The duration of acceleration will be dependent upon the crewman's judgement of the distance to be traveled. The S.M.U. provides attitude stabilization and will automatically maintain the crewman's orientation in space. As the crewman progresses toward his target object he will move above or below it instead of directly toward it due to his change in velocity and orbital altitude. To correct this the crewman will use the up or down translation control. By using this technique he will follow a damped oscillatory path to the target object, the magnitude of the oscillations being dependent upon the crewman's capability to recognize his drift above or below the target. Once the crewman is close to his target object he will give himself an aft acceleration to slow himself down to zero when he reaches the target. Here again the duration of the acceleration is dependent upon the crewman's judgement.

The valves which control the energy impulses to give orientation and translation control are "on-off" valves. The controls therefore are pulse controls, the duration of the pulse determining the amount of energy released.

1.3 PROBLEM AREAS REQUIRING FURTHER INVESTIGATION

1. Assembly technique to be used for assembling the space vehicle; Remote control vs. Direct Control.
2. Transfer technique to be used; Docking vs. Close Approach.
3. Air Lock Design
4. Space Suit Requirements Design
5. Emergency Requirements
6. Maneuvering Unit Design
7. Influence of an Orbital Launch Facility on Crew Transfer

1.4 CONCLUSIONS AND RECOMMENDATIONS

The Apollo missions could be accomplished sooner without affecting crew safety during launch into earth orbit by utilizing the crew transfer approach.

Studies should be continued in the problem areas mentioned above in order to define the impact on the Apollo space vehicles and the earth launch vehicles, of crew transfer for the orbital launch approach.